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VOYAGER SPACECRAFT Phase B, Task D

FINAL REPORT

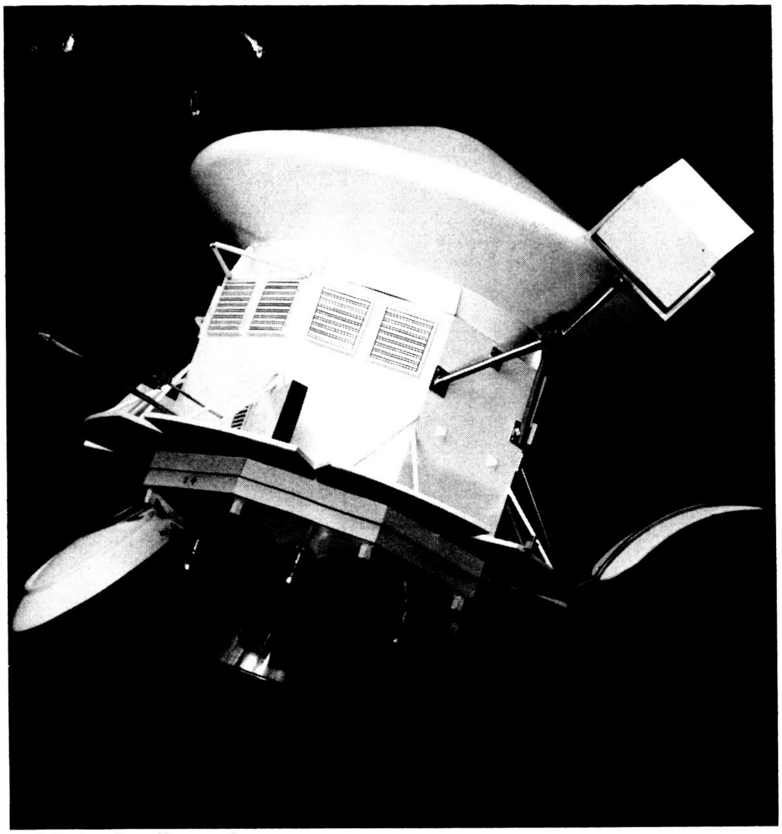
OCTOBER 1967

Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

Voyager Operations
Space Vehicles Division

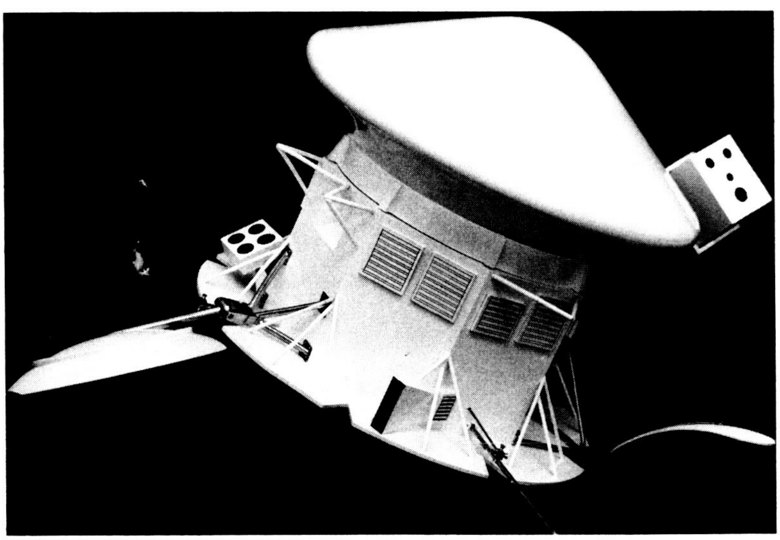
TRW SYSTEMS GROUP

One Space Park, Redondo Beach, California

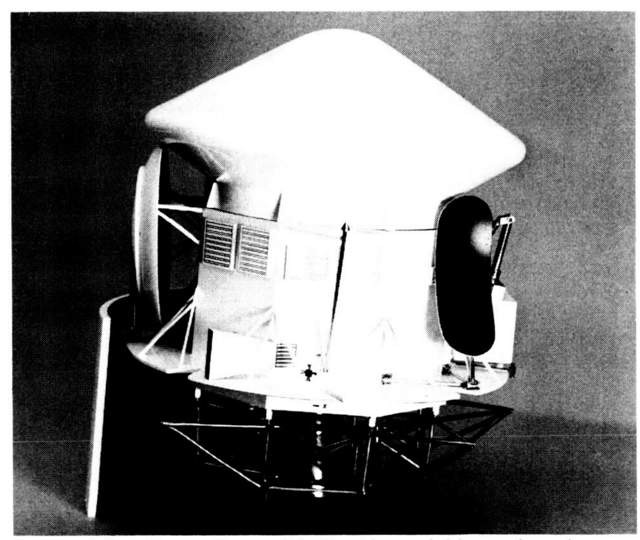


In-Flight Configuration

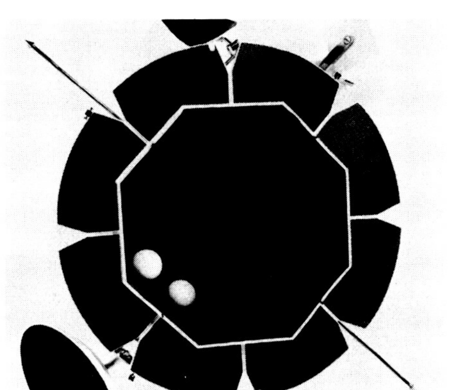
MODEL OF
TRW
RECOMMENDED
VOYAGER
SPACECRAFT



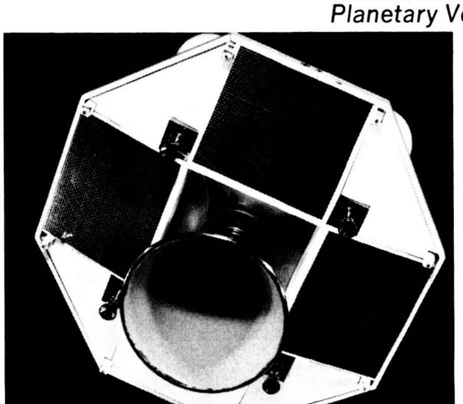
Opposite View In-Flight Configuration



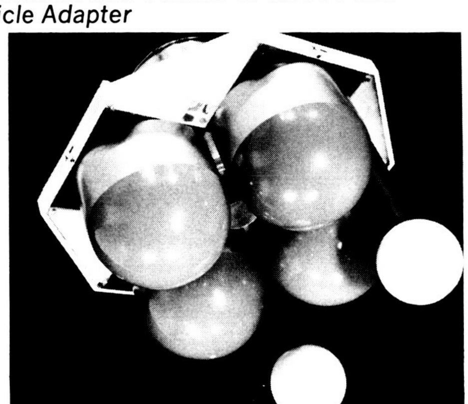
Stowed Configuration with Section of Shroud and Planetary Vehicle Adapter



Propulsion Module, Top View



Propulsion Module, Bottom View



Equipment Module, Bottom View

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VOYAGER

SPACECRAFT Phase B, Task D

FINAL REPORT

Volume 7. Preliminary OSE and MDE

OCTOBER 1967

Prepared for
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INTRODUCTION

The Voyager Phase B, Task D study contract called for identification of the preliminary requirements for the MDE and OSE necessary to support the Voyager spacecraft defined in the study.

Volume 7 contains the results of this preliminary requirements study and in addition provides a preliminary description of support equipment and computer programs involved. Many of the electrical hardware items are common to both MDE and EOSE. Complete descriptions of the common items are contained in Sections 6 and 12. The mechanical OSE is described and illustrated in Sections 16 and 17.

The technical descriptions in this volume, when combined with the implementation sections in Volume 8 for spacecraft launch operations, flight operations, and logistics, comprise the complete Task D study results with respect to spacecraft systems integration and test activities.

One of the specific Task D studies assigned to TRW was to investigate various candidate equipments for spacecraft imaging subsystems. While the results of this study are contained in Volume 11, the ground image reconstruction equipment for use with these spacecraft systems is described in this volume along with MDE hardware descriptions in Section 6.

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ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

1. MDE HARDWARE INTRODUCTION

The mission-dependent equipment (MDE) consists of all the specialized equipment required to support the projected 1973 through 1979 Voyager mission requirements of the Deep Space Networks (DSN), including its 210-, 85-, and 30-foot antenna Deep Space Stations (DSS) and Space Flight Operations Facility (SFOF) as well as the Huntsville Operations Support Center (HOSC). The MDE system is designed to make maximum use of present and planned DSN and HOSC equipment, software, operational philosophy, and procedures. Where information on planned DSN configurations, such as defined in JPL Document No. EPD-283, indicates potential improvements, flexibility has been incorporated into the MDE system to include these planned features.

The spacecraft MDE design is based on sharing the mission-independent equipment (MIE) capabilities of the DSN with capsule and surface laboratory MDE such as to result in an integrated and balanced system capability. The system is sized for simultaneous operation of two Mars orbiting spacecraft and two capsule payloads. The system is also designed to meet operational requirements during the high-activity mission modes associated with simultaneous vehicular operation during periods of interplanetary maneuvers, Mars orbit insertion, capsule separation, and capsule entry and landing. When the MDE is integrated with the MIE at the DSN and HOSC, a ground-operational system will exist capable of supporting the tracking, telemetry, data processing and display, and command generation and verification requirements of the Voyager spacecraft system. Figure 1-1 shows a conceptual block diagram of a typical DSS depicting the integrated in-line Voyager MDE/MIE, their interfaces and planned data flow.

In addition to the in-line MDE shown in Figure 1-1, the MDE includes special operational equipment required for DSN subsystem and system readiness testing and training and for verification of the performance capabilities of the MDE/MIE at threshold conditions. Simulation equipment and computer programs are provided as necessary for operator training.

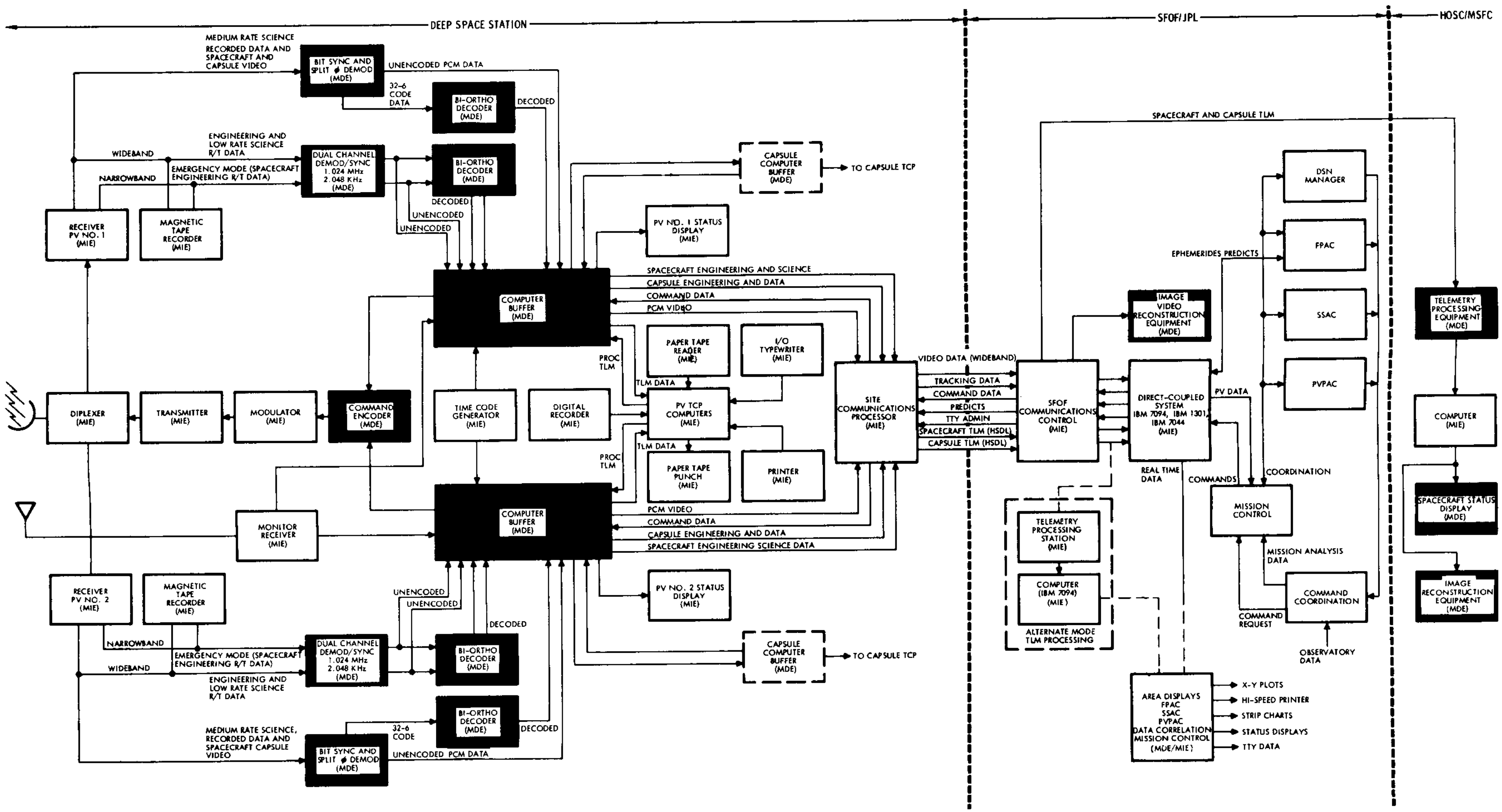


Figure 1-1
DSN CONCEPTUAL COMMAND AND DATA PROCESSING SYSTEM, depicting in-line Voyager MDE/MIE, their interfaces and planned data flow from The Deep Space Stations to the Space Flight Operations Facility and Huntsville Operations Support Center.

Based on existing NASA planning documentation, MDE will be provided to DSS 12 and 14 (Goldstone, California), DSS 42 and 43 (Canberra, Australia), DSS 62 and 63 (Madrid, Spain), DSS 71 (Kennedy Space Center, Florida), and DSS 72 (Ascension Island) to support the Voyager missions. MDE will also be provided to the SFOF at Pasadena which will be the central point for control of mission operations and to the HOSC at the Manned Space Flight Center, which will perform mission support functions.

Contained in the ensuing paragraphs are the following MDE sections, scoped as indicated:

- Mission Operations Complex. This section describes the mission operations complex (MOC) system requirements for support of the 1973 through 1979 Voyager mission required at the Air Force Eastern Test Range (AFETR), the DSN, and the Manned Space Flight Network (MSFN) from launch through completion of mission operations. MOC system operating modes, data processing requirements, computer programs, command links, and system test requirements are described.

- MDE/MIE Functional Equipment. Functional requirements and interrelationship of the DSN and HOSC MDE/MIE in-line equipment required to support the Voyager mission.

- MDE Operational Test and Checkout Equipment. The MDE test and checkout equipment and its functional requirements and interfaces with the in-line MDE/MIE.

- MDE Configuration. Description of the MDE hardware configuration in terms of numbers of racks and cabinets required and the physical arrangement of the assemblies and racks which comprise each of the MDE subsystems.

- MDE Hardware Description. Design performance and interface requirements for each of the in-line items of MDE are defined.

2. MISSION OPERATIONS COMPLEX

Voyager mission support will be provided by a mission operations complex (MOC) comprised of existing and planned capabilities of the DSN and other NASA and Department of Defense facilities, augmented with MDE as appropriate. These facilities will include:

- General purpose digital computing facilities at selected Deep Space Instrumentation Facilities (DSIF), Air Force Eastern Test Range (AFETR), Manned Space Flight Network (MSFN) stations, and the SFOF and HOSC for real-time and off-line handling of Voyager operational data
- NASA Communications System (NASCOM) data circuits to handle tracking, telemetry, video, and command data between data acquisition stations and the SFOF and voice and teletype nets as required
- Physical accommodations and facilities in the SFOF for mission operations system teams
- Standard display and computer remote input-output equipment for mission operations team for command and control of Voyager operations at SFOF and for operational support at HOSC.

Equipped with these facilities, the MOC will perform the following functions in support of the Voyager mission described herein:

- Track the space vehicles and provide metric tracking data
- Receive, record, process, and display telemetry data from the space vehicles
- Transmit commands generated at the SFOF to the space vehicles
- Provide station performance parameters which are required for analysis and evaluation of vehicle performance
- Provide and maintain a library of master data records developed during each flight
- Provide acquisition data required by tracking and data acquisition stations.

From launch through interplanetary injection, the MOC will provide metric tracking coverage and telemetry data acquisition coverage for the space vehicles. This includes tracking and telemetry data acquisition coverage throughout the earth parking orbit phase. After interplanetary trajectory injection, approximately two hours of S-IVB metric tracking coverage will be required for S-IVB orbit determination.

For 30 days from interplanetary trajectory injection, the MOC will provide, within view capabilities, continuous DSN metric tracking coverage, telemetry data acquisition coverage, and command coverage for each planetary vehicle. Approximately one-half hour of overlapping metric tracking coverage is required when a planetary vehicle is within view of two DSN sites.

From interplanetary trajectory injection plus 30 days until Mars encounter minus 20 days, the MOC will provide continuous telemetry data acquisition coverage and command coverage for each planetary vehicle and 12 hours of continuous metric tracking coverage every two days for each planetary vehicle during cruise. During each period, approximately one-half hour of overlapping metric tracking coverage will be required when a planetary vehicle is within view of two DSN sites. For planetary vehicle interplanetary trajectory corrections, the system will provide five days of continuous metric tracking coverage prior to the correction and 10 days continuous metric tracking coverage after the correction.

From Mars encounter minus 20 days until spacecraft-capsule separation, the MOC will provide continuous metric tracking coverage, telemetry data acquisition coverage, and command coverage for each planetary vehicle. From spacecraft-capsule separation until the termination of Mars orbital operations, the system will provide continuous telemetry data acquisition and command coverage for each flight spacecraft. In addition, the system will provide continuous metric tracking coverage of every other orbit plus continuous metric tracking coverage during occultation experiments for each flight spacecraft.

Specific MOC Voyager support requirements for the 1973 period are described in the following paragraphs. In determining MDE requirements of the mission operations complex for the Voyager mission, TRW's experience with the AFETR, DSN, and MSFN gained from spacecraft programs such as OGO, Vela, Pioneer, and Apollo was utilized. Also, JPL's Engineering Planning Document (EPD) Number 283, dated 1 January 1967, and other supplementary NASA information were used to determine what the planned capabilities of the DSN are for Voyager 1973. The EPD-283 document, in particular, represents an integrated source of information concerning the estimated DSN configuration and provided a useful means of assessing the MIE ground systems capabilities planned for spacecraft flight support. Where specific Voyager MOC requirements cannot be supported by the planned DSN configuration, analyses and tradeoff studies were conducted to determine the optimum method of fulfilling the requirement through MDE equipment or software.

2.1 MOC FACILITIES

2.1.1 Air Force Eastern Test Range

The Air Force Eastern Test Range, which comprises a part of the MOC for Voyager, will track the launch vehicle, receive telemetry from the launch vehicle and the two planetary vehicles, and provide data handling support during the near-earth Voyager operations. Instrumented aircraft, ships, and range stations will track the vehicle from launch to provide metric and telemetry data. These aircraft, land, and ship-based instrumentation systems will be linked by a communications system with Kennedy Space Center (KSC), SFOF, and HOSC during near-earth operations.

Pointing information and other predict data acquisition by the AFETR will be sent to the DSN acquisition stations. TRW's review of the AFETR capabilities versus Voyager tracking accuracy and ranging requirements indicates that at this time, the station's MIE is adequate. Therefore, no MDE is currently planned for the AFETR.

2.1.2 Deep Space Network

The DSN has the capability for two-way communications with, and has the tracking and data-handling equipment to support, unmanned space vehicle operations at earth-referenced distances greater than 10,000 miles. The main elements of the DSN are the Deep Space Instrumentation Facility, the Ground Communications System, and the Space Flight Operations Facility.

2.1.2.1 Deep Space Instrumentation Facility

The DSIF will utilize the following S-band stations for Voyager:

- The planetary vehicle monitor station, Cape Kennedy, will be used for spacecraft-capsule-DSN compatibility verification and for telemetry reception from liftoff until the end of the viewing period.
- A network of three 85-foot antenna stations will be used for coverage from planetary vehicle injection to near-planetary encounter and for backup during later mission phases. The specific stations will be Madrid, Spain; Canberra, Australia; and Goldstone, California.
- A network of three 210-foot diameter antenna stations will be used for coverage during the later phases of transit and orbit and landed Mars operations. These stations also are at Madrid, Canberra, and Goldstone.

The Spacecraft Command and Guidance Station, Ascension Island (30-foot-diameter antenna), will be used for acquisition, if visible, and for near-earth command transmission, telemetry reception and tracking.

Spacecraft range measurements at the DSS are related to the time difference between two identical, separately generated, pseudo-random signals, one generated at the station transmitter and phase-modulated on the carrier, and the other generated at and synchronized by the station receiver for correlation detection. The transponder in the Voyager spacecraft will receive the transmitted signal and retransmit the same modulation in a "turn-around" mode back to the interrogating DSIF station. A turn-around ranging system, capable of being used for precision station-to-station time synchronization to within a few microseconds, exists throughout the DSIF at the present time. Planetary ranging equipment with a noncoherent clock will be available at the 210-foot stations. A noncoherent clock allows a ranging fix without first locking the doppler system.

Two-way doppler data will be used at the DSS to obtain tracking data for orbit determination purposes. The technique will involve transmitting a precision carrier to the spacecraft, where it will be coherently shifted and sent back. The ground receiver will then compare the phase of the received carrier with that of the transmitted carrier to extract the doppler data.

A general-purpose tracking data handling system computer will be used to sample and format tracking data for transmission to SFOF. The subsystem will provide programmable sample rates, integration times, and information for the following items: space vehicle ID number; data conditions, Greenwich Mean Time; antenna hour or azimuth angle; antenna declination or elevation angle; doppler frequency, range data, including range condition code; transmitter frequency; and the day of year. This subsystem will be capable of handling two simultaneous, independent tracking data streams. Software, sample rates, mode of operation, ID assignments, and monitoring and validation of tracking data will be provided by the DSN. No MDE is required to support the DSN tracking operations.

2.1.2.2 Ground Communication System

The present Ground Communication System (GCS) is a part of NASCOM and will be based on existing 1973 planning data, capable of providing the additional facilities and equipment required for an integrated network. Teletype communications will be provided between all overseas tracking and data acquisition stations (85 and 210 feet) and various computation and control centers. Voice link capabilities will include telephone and four-wire, nonsignaling conferencing networks within and between the DSIF stations and SFOF. The network will also include high-speed data circuits for information transfer at various rates using standard data-conditioned channels.

The GCS will utilize communications-oriented computers (communications processors) which automatically read routing information within a given message and, on the basis of this information, switch the message to its proper destination.

In order to facilitate automated switching, the high-speed data stream will be broken into uniform data blocks. Each block will contain the following information: sync words, source code, destination code, data ID, data, and an error detection code.

Communications satellites and underseas cables will provide reliable low-error-rate communications during the Voyager operational era. No MDE is planned for the ground communication system. However, the design of the MDE for use with the integrated DSN system is based on the assumption that video data will be handled in non-real time between the DSIF stations and SFOF and HOSC using the 50,000-bit wide band data link as specified in JPL EPD 283. Telemetry data will utilize the standard high-speed data links (HSDL) and teletype circuits.

2.1.2.3 Space Flight Operations Facility

The SFOF will house a central complex providing the means by which the mission, the spacecraft, and the DSN can be controlled and operated. The purpose of the SFOF is to provide for data processing, analysis, display, communications, and support which may be used in conjunction with the DSIF for rehearsing and executing Voyager space-flight activities.

In its present configuration, the SFOF has four major elements: data processing system, support system, DSN communications system, and simulated data conversion center. Although mission-dependent equipment will be provided to the SFOF to support the Voyager activities, there are no requirements for MDE in relation to spacecraft tracking operations other than software, which is defined later in this volume.

2.1.3 Goddard Space Flight Center/Manned Space Flight Network

Because of DSN acquisition limitations for spacecraft at altitudes less than 10,000 nautical miles, the combined coverage afforded by the AFETR and the DSN will require supplementation by selected MSFN stations from liftoff to planetary vehicle injection. This supplementary coverage will require the acquisition of spacecraft and capsule engineering performance data. Since these data are on 1.024-mHz subcarriers at a bit rate of 512, it is possible to demodulate this data using equipment similar in design to the unified S-band demodulator presently at these sites. These data will be recorded and, if required, baseband transmission to SFOF or HOSC in real time can be accomplished. There are no MDE requirements for the MSFN tracking operations.

2.2 MOC OPERATING MODES

The Voyager ground data handling system is designed for use in various operating modes since the data handling requirements, especially for real-time or quick-look data, will vary widely during different phases of the mission. For periods of high operational activity, such as during midcourse maneuvers, planetary encounter, capsule separation, and landing, the MDE is designed to make use of the maximum DSN data handling capability. During periods of sustained low-level activity, such as cruise mode operations, extended periods of routine scientific exploration, or subsequent to completion of major mission objectives, the MDE is designed to operate with minimal use of the DSN mission-independent equipment. Thus the MDE is designed to operate in a mode for transferring maximum data to the on-line data processing system at the SFOF via high-speed data lines (HSDL) when required, and to use an alternate mode for low-activity periods. In the latter mode, the telemetry processing station at the SFOF or the telemetry and command processors (TCP) at the DSS can perform the essential functions of decommutation, processing, and driving displays at the SFOF for the various user areas. In this mode the major part of the data will be recorded for off-line reduction and analysis. This degree of flexibility can be simply achieved through switching equipment configurations within the DSS and ground communications system and by loading different programs into the TCP computers. The use of the reduced-level operating mode will minimize equipment conflicts during periods of multiple project operations and will provide considerable savings in operational costs during extended periods of low activity.

2.3 MOC DATA PROCESSING

As shown in Figure 1-1, the spacecraft engineering, science, and video data is received by the S-band receivers at the DSS and the receiver output, consisting of separate subcarriers and split phase telemetry data, is fed to the telemetry demodulator. The data is then demodulated into a PCM bit stream in a NRZL format, along with bit-rate clock signal and synchronization status information. Depending upon whether the data transmitted from the spacecraft is uncoded or biorthogonally encoded, the output of the demodulator is sent under TCP computer control either directly to the computer buffer or first to the biorthogonal decoder where the coded word is determined and then forwarded to the computer buffer. Under control of the TCP computer, the buffer transfers the uncoded or decoded PCM data to the computer in parallel groups of preset size. The data is then processed and routed via the buffer in serial form to the DSS monitor and status displays and to the site communications processor for transmittal in near-real time to the SFOF and HOSC. In addition, the TCP computer identifies blocks of capsule telmetry or video data, which may be relayed on the orbital spacecraft downlink, and routes this data to the capsule TCP computer, where the data is decommutated and processed under control of the capsule data processing program.

Telemetry data acquired at the Deep Space Station, depending upon the data rates, priorities, and mode of operation, will be transmitted to SFOF and HOSC by teletype (TTY) or high-speed data lines (HSDL). Video data, depending upon data rates, will be transmitted to SFOF and HOSC via HSDL or wideband data link. Image reconstruction of video data will be performed at the SFOF and HOSC, but not at the deep space stations.

2.4 MOC COMPUTER PROGRAMS

The TCP computers at the Deep Space Stations will be variously used, depending on whether the bulk of the received data is being transmitted directly to the SFOF for entry into the on-line computer facility or whether the TCP computer is performing a data processing function independent of SFOF computers. Functions that will be performed by the TCP computer in the various operating modes include:

- Synchronize data
- Decommutate data
- Recognize, switch, and route data blocks, frames, or words
- Drive local operator displays with selected spacecraft parameters
- Access time from the local station timing system
- Time-tag telemetry data messages
- Accept inputs from a local command monitor-receiver to check the transmission of command messages and inhibit transmission of incorrect commands
- Format telemetry data into engineering units for teletype transmission to users
- Generate alarm signals or typewriter printout for prescribed alarm situations.

The computer programs to be implemented in the SFOF will process and correlate incoming data so as to display information to the science, flight path, and engineering analysis teams in the most useful form possible. The mission-dependent operations computer programs will be integrated into the system with existing mission-independent and mission-dependent programs. The Voyager computer system will retain the capability to run certain independent programs off-line to support multiple operational functions simultaneously or for backup of critical functions during maneuvers and periods of intense mission activity.

In addition to the on-line operational computer programs and off-line analysis programs, simulation, diagnostic, and other test computer programs will be provided to facilitate the software integration checkout and certification process and to develop the simultation data required for a flexible personnel training and operational test program. It is a design goal to develop computer programs that will generate DSN simulation data for any one of a number of nonstandard operations sequences. These computer programs will furnish training and simulation data for rehearsal exercises to help operations personnel identify nonstandard operations, diagnose nonstandard operations, and learn to make correct operational decisions. A detailed description of these programs is provided later in the MDE software section of this volume.

2.5 MOC COMMAND LINKS

The command system will provide a capability to load pre-programmed commands into the programmer of either orbital spacecraft or capsules in quick succession through automatic transmission of commands generated in the SFOF computer. An alternate command capability will exist for manual generation of commands by a command encoder at the DSS when this mode of commanding is directed from the SFOF.

The Voyager spacecraft command system is capable of accepting time-tagged commands for storage in the computer and sequencer. In the normal mode of operation, the uplink carrier is modulated with a phase-shift keyed (PSK) subcarrier containing digital data at a command rate of eight bits/sec. Groups of commands are generated by the SFOF computer and automatically processed through the DSS station MDE command encoder. In the manual command mode, the resulting command is also transmitted at eight bits/sec using the same subcarrier frequency.

Prior to transmission of automatic commands, the correct command format is entered into the TCP computer and held in memory for comparison with the actual radiated command format. During transmission a bit-by-bit transmission check is made on each command. The radiated command is detected by a command monitor receiver, which feeds the signal corresponding to each radiated data bit via the computer buffer to the TCP computer. In the event of an incorrect bit, further transmission of the command is inhibited. The TCP computer will also contain a permissive command list against which all commands will be compared as a check prior to the beginning of their transmission. The appropriate permissive command list is loaded into the computer for each phase of the mission to preclude inadvertent generation of catastrophic commands.

Command verification is performed either by the DSS computer or by the SFOF computer, depending upon which mode of operation is in use. Figure 2-1 shows a simplified block diagram of the Voyager command data flow.

The command system is provided with all necessary safeguards to ensure, insofar as possible, that command operations during actual

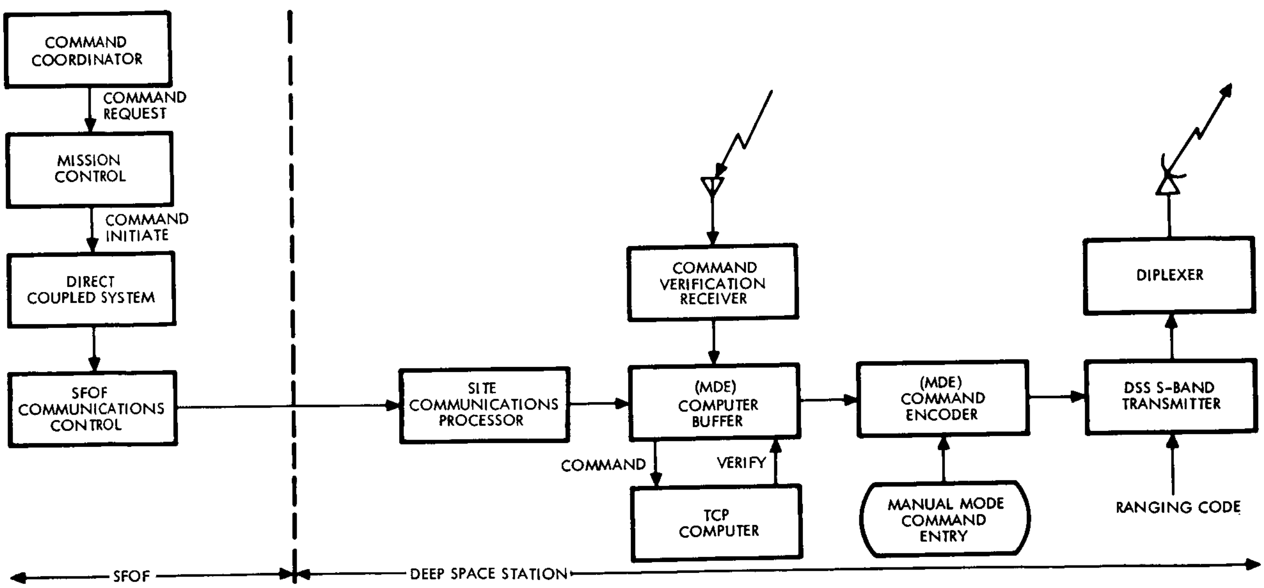


Figure 2-1
COMMAND DATA FLOW provides for SFOF control and verification of all spacecraft commands sent either automatically or manually.

mission operations are strictly under the control of the space-flight operations director at the SFOF. However, provisions for command monitoring and equipment for checkout and verification of proper operation of the command system are provided for use by local station personnel.

2.6 MOC SYSTEM TESTING

The MDE is designed to accommodate patchboard switching at the Deep Space Stations, to maximize the capability for off-line checkout and validation with minimum use of mission-independent equipment. Digital recording will be used to the maximum extent to eliminate patching and calibration of analog recorders. Telemetry and command computer programs will provide for standard interfaces with communications processors, digital instrumentation systems, and station displays and timing subsystems.

Tests of mission operation equipment and personnel training at levels beneath the full system tests will afford a capability for attaining high levels of confidence without excluding DSN facilities from the support of other flight projects. This capability will be implemented by telemetry simulation programs at both the SFOF and at the stations for personnel training and limited equipment checkout. The use of MDE communication test and simulation tapes and RF test transponders and simple simulation devices during subsystem test and checkout of mission-dependent equipment will prevent excessive utilization of general purpose equipment for test and training. As an example, simulation by the command encoder of the output of the monitor receiver to the computer buffer will preclude a signal input from the command monitor receiver during isolated testing of command operations.

3. MDE/MIE FUNCTIONAL EQUIPMENT

The MDE consists of in-line operational equipment, test operational equipment, and standard test equipment. The in-line MDE is comprised of the following subsystems which include the functional elements shown in Figures 3-1 through 3-3.

- Telemetry and data processing subsystem
- Biorthogonal decoding subsystem
- Command generation subsystem
- Image reconstruction subsystem
- Planetary vehicle subsystem display.

The MDE in-line hardware subsystems are interconnected with the DSN MIE by the MDE cable sets.

The functional relationship of the deep space network and HOSC MDE/MIE in-line equipment is depicted in Figure 1-1. Functions performed by the MDE are as indicated, namely, command generation,

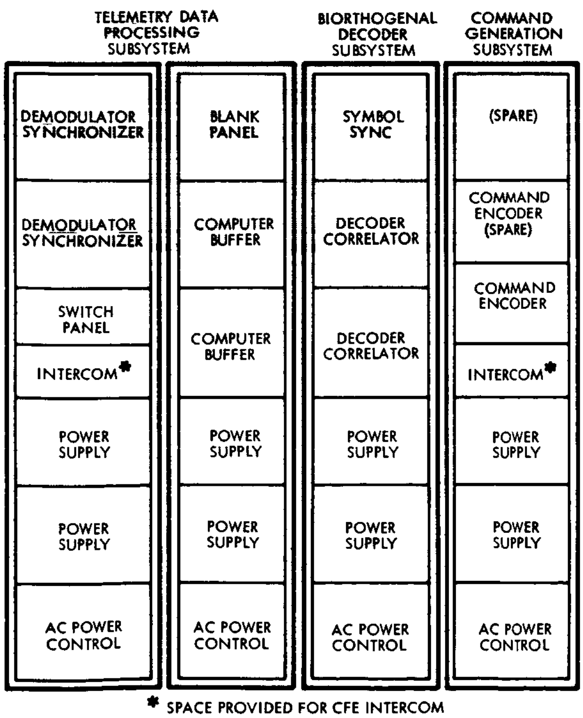


Figure 3-1
MDE IN-LINE EQUIPMENT AT DEEP SPACE STATION consists of the Telemetry Data Processing Subsystem, Biorthogonal Decoder Subsystem and Command Generation Subsystem.

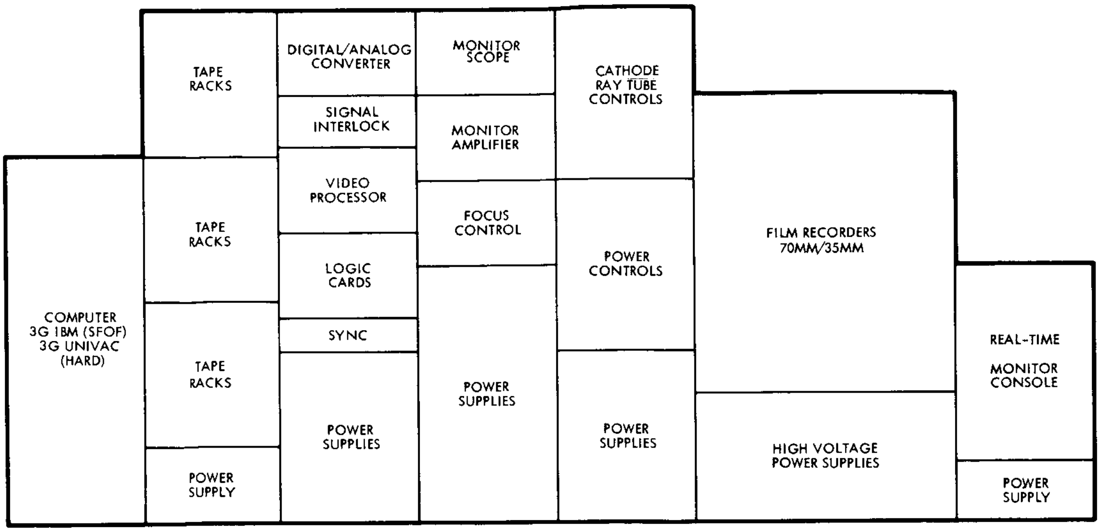


Figure 3-2
MDE IMAGE RECONSTRUCTION EQUIPMENT will be provided to the SFOF and HOSC but not the DSS to process and reconstruct the spacecraft video data.

data demodulation and decoding, computer buffering, data processing, image reconstruction, and planetary vehicle status display. Image reconstruction of spacecraft television or photographic data will be done at the SFOF and HOSC but not at the DSS. System requirements of the in-line MDE/MIE are as follows.

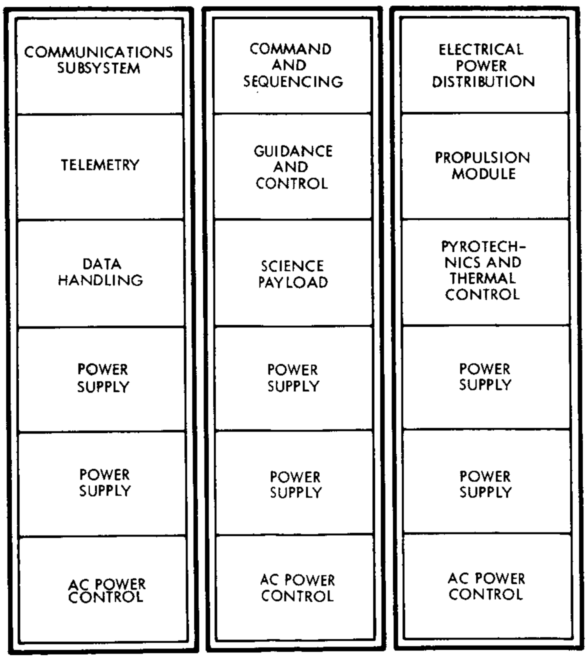


Figure 3-3
MDE AT SFOF AND HOSC includes the Planetary Vehicle Subsystem Display to augment existing computer driven displays.

3.1 TRANSMITTER/RECEIVER (MIE)

The S-band system at each DSS will be capable of handling two coherent two-way channels simultaneously with noncoherent receiving capability on two additional channels. This configuration provides a capability during periods of maximum activity for simultaneous tracking of the two orbital spacecraft while at the same time receiving telemetry data from both spacecraft and both capsules.

The stations incorporate sensitive and stable telemetry receivers that are designed to track the received 2300-MHz carriers and detect both amplitude and phase modulation. The telemetry subcarriers, derived from the appropriate detection channels, will be parallel routed to magnetic tape recorders (MIE) and the mission-dependent telemetry demodulators. There are no MDE requirements for tracking equipment at the DSN or HOSC.

3.2 TELEMETRY DEMODULATOR (MDE)

The signal inputs to the telemetry demodulator from the DSS receiver will consist of three data channels. One channel, containing planetary vehicle Link 1 telemetry data, will consist of a 1.024-MHz subcarrier modulated with real-time engineering data. A second channel used during emergency operating mode will contain planetary/ vehicle Link 2 engineering data modulated on a 2.048-kHz subcarrier. The third channel will contain planetary vehicle Link 3 split phase telemetry data comprised of spacecraft and capsule video, recorded engineering and science data, and real-time capsule engineering data. The telemetry signals of all three channels contain both planetary vehicle data and synchronization information in either a 32, 6 biorthogonal code or 6-bit uncoded words, depending upon the nature of the signal transmitted from the planetary vehicle. The primary outputs of the demodulator will be detected coded data or uncoded reconstructed serial pulse coded modulation (PCM) data bit streams in a NRZL format. All three channels will contain bit-rate clock signal synchronization and synchronization status information. Outputs from the demodulator synchronizers will be decoded at the DSS prior to sending the information to the communication processor for transmission to the SFOF/JPL and HOSC/MSFC. Contained in Table 3-1 is a summary of the spacecraft communication downlink data transmission parameters.

3.3 BIORTHOGONAL DECODER (MDE)

All coded data from the demodulator/synchronizer will be fed directly to the biorthogonal decoder. Upon receipt of the clock and integrated data symbols plus noise from the demodulator/synchronizer the decoder will perform a correlation detection process whereby each symbol of the coded word will be quantized and vectorially multiplied by each of the probable transmitted 32 symbol coded words and its complement. An algorithm will then be performed for each of the possible 32 symbol code words transmitted. The resulting values will be stored in accumulators which, at the end of a 32 symbol code word, will be compared. The largest value accumulated corresponds to the six-bit signal transmitted from the spacecraft. Thus, the output of the decoder will be comprised of six-bit unencoded PCM serial bit stream words and synchronization data.

3.4 COMPUTER BUFFER (MDE)

The computer buffer will serve as a central point of distribution for inputs to the telemetry and command processor (TCP) for planetary vehicle data and the site communications processor from the biorthogonal decoders or the telemetry demodulators, the command encoder, and the command monitor receiver. Under the TCP computer control, the buffer will transfer serial PCM data to the computer in parallel groups of preset size. Similarly, it will route this processed telemetry data from the computer in serial form to the DSS monitor and status displays and to the site communications processor for transmittal in near-real time to the SFOF and HOSC.

3.5 TELEMETRY AND COMMAND PROCESSOR (MIE)

The telemetry and command data subsystem, assumed to comprise the third-generation Scientific Data Systems computer or equivalent, will provide for on-site telemetry and command data processing. This computer is assumed to have an 850-nanosecond memory cycle and changeable core memories expandable in 4096 32-bit word

Table 3-1. Planetary Vehicle Communication Downlink Transmission Data*

Link	Mode	Modulation	Type Information	Data Rates	Symbol Rates	Subcarrier Frequency
1		PSK/PM	Planetary vehicle engineering and low-rate science real-time data	512 (NRZL)	2.73 kb/sec	1.024 MHz
2	Emergency	PSK/PM	Spacecraft engineering real-time data	8 (NRZL)	42.6	2.048 kHz
3	1	Split PM	Spacecraft video, capsule video, recorded engineering, and medium engineering rate science data	51.2 kb/sec** 25.6 kb/sec** 12.8 kb/sec** 6.4 kb/sec*** 3.2 kb/sec***	273 kb/sec 136.5 kb/sec 68.25 kb/sec 34.125 kb/sec 17.063 kb/sec	273 kHz 136.5 kHz 68.25 kHz 34.125 kHz 17.063 kHz
3	2	Split PM	Capsule entry real-time engineering data	512	2.73 kb/sec	2.73 kHz

Split phase

* Planetary vehicle communication downlink transmission data are summarized for Links 1, 2, and 3 as to type of information, data and symbol rates, and subcarrier frequencies.

** Encounter to encounter + 6 meters.

*** Failure mode.

increments to 131,072 words. By employing the dual computer capability shown in Figure 1-1, the TCP system will be capable of performing all necessary functions of near-real-time data processing for each of the planetary vehicles. Each of the computers will be programmed for processing planetary vehicle engineering, science, and video data. Demodulated telemetry data from the demodulators or biorthogonal decoders will be processed by the computer and transferred via the computer buffer to the site communications processor for transmission to the SFOF and HOSC. Capsule data will be routed via the computer buffer to the capsule buffer. All data processed by the computers will be recorded on magnetic tape. Functions to be performed by the TCP computer in conjunction with computer software (MDE) are:

- Data synchronization
- Selective editing of spacecraft and capsule telemetry data
- Generate alarm signals or typewriter printout
- Decommutation of telemetry data for local displays
- Drive local operator displays with selected planetary vehicle parameters
- Formating and time coding of telemetry data for transmission to the SFOF via TTY, high-speed data line or wide-band data link.

3.6 DSS DISPLAYS (MIE)

Several telecommunications parameters will be displayed in engineering units at the deep space stations. The displays (MIE) will be computer driven. As a minimum, the parameters displayed will include spacecraft receiver static phase error, received signal level, power output, and command verification. The SFOF and HOSC computer driven displays will be augmented by a planetary subsystem display (MDE) (Figure 3-3) and planetary vehicle mission display (MIE). The former will present information relative to operating modes, events, malfunctions, and significant engineering parameters for each of the planetary vehicle subsystems. Mimic lines depicting significant functions of each spacecraft subsystem will be displayed on the front panels of this equipment. The planetary vehicle mission display will include general purpose mission status displays, computer driven to indicate planetary vehicle mission progress. One display will indicate the relative geometry, dynamically, of earth, Mars, the sun, Canopus, the two orbiting spacecraft about Mars, and the location of the capsules on the surface of Mars. This display will indicate to operational personnel at any time the possible combination of communications links available as well as the impending visibility paths and occulations. Overall vehicle status and a mission event profile will be shown on separate parts of the mission display.

3.7 SITE COMMUNICATIONS PROCESSOR (MIE)

A communications processor will be provided at each Deep Space Station for circuit routing and for system monitoring functions. It will insert and extract NASCOM message preambles, recognize and return NASCOM circuit assurance messages, keep a message count, and perform coding and decoding for one duplex error-correcting command channel. The ground communication system will handle four 100 word per minute circuits, four 2400 bits/sec high-speed data circuits, one 50-kb/sec wideband data link, and four voice circuits in support of Voyager operations.

The subsystem consists of a general purpose computer with appropriate peripheral data communications equipment to handle the TTY and high-speed data inputs and outputs. Additional peripheral equipment is two buffers to transmit communications channel error counts and self-check (diagnostic) information to the station's digital instrumentation system, a TTY keyboard send-receive unit for communications operator control, and a TTY receive-only page printer for message logging.

3.8 SFOF DATA PROCESSING SYSTEM (MIE)

The SFOF data processing system is a complete, integrated, operating computer system to meet the requirements of several flight projects for simultaneous flight operation support by the DSN. For the Voyager mission operation this system will be used with an MDE executive program which will provide sequential calling of appropriate programs and routines.

3.9 TELEMETRY PROCESSING STATION (MIE)

After entry into the SFOF, high-speed telemetry data may be processed in the telemetry processing station as a backup to the data processing system. Functions performed will be:

- Convert received telemetry data to a 36-bit parallel format compatible with a 7288 high-speed subchannel and to IBM compatible magnetic tape
- Provide the capability for off-line analog data analysis
- Provide the capability for producing strip chart recordings of analog data outputs
- Provide the capability for recording all composite and high-speed digital data entering the SFOF via wideband data link, and other DSIF high-speed data sources.

The conversion process will be in real time, using signals received from the stations or in non-real time using data recorded on magnetic tape. The intent of the use of the telemetry processing station is to minimize special purpose equipment as well as to obviate the need for the data processing system computer over extended periods of time.

During critical portions of the mission the station will provide parallel processing, thus assuring a backup in the event of failure of the prime processing path.

3.10 RECORDING EQUIPMENT (MIE)

All planetary vehicle telemetry and command signals which pass through the RF system will be recorded by FR 950 or FR 1400 analog tape recorders. Also, digital recordings of tracking data, station performance, and processed telemetry and command signals will be made using MIE available at the DSS. Both the analog and digital tapes will be used if required at the SFOF and HOSC for non-real time image reconstruction and data analysis.

At each station, digital recorders associated with the TCP will be provided as MIE by the DSS. Similarly, strip chart and magnetic tape recorders required at the SFOF and HOSC will be provided as MIE.

3.11 COMMAND ENCODER (MDE)

The command encoder will provide for either automatically or manually encoding commands transmitted from the DSS to the planetary vehicle. The resulting command from the encoder will be a phase-shift keyed, square wave subcarrier signal which will be used to modulate the 2100-MHz station transmitter frequency. The command encoder transmission rate will be eight bits per second and will be categorized as either discrete (execute immediately), stored -program (execute immediately), or stored-program (store in spacecraft computer and sequencer) commands. Simultaneous commanding or commanding and ranging of the spacecrafts or the capsules is not required. The command encoder will be capable of addressing two spacecrafts plus a spare spacecraft for backup to launch operations and either of two decoders contained in a vehicle. In addition, the command encoder will be capable of generating time tags for test purposes and emergency operations. Specific requirements for each of the two command encoder operating modes are as follows:

- Normal Mode. In the normal mode, commands automatically generated by the SFOF computer and transmitted to the DSS telemetry and command processor (TCP) will be serially entered into the command encoder automatically and transmitted to the spacecraft after verification checks have been performed. Prior to the command encoder transmitting a command, the message will be formatted by the telemetry and command computer in accordance with the data received from the command generation computer at the SFOF via the DSS communication processor. The potential command will be held in memory by the DSS/TCP computer for comparison with the radiated signal during transmission.

  The normal mode of commanding the spacecraft will be as follows:

  Coordination of all command requests from the Space Science Analysis and Command (SSAS), Planetary Vehicle Program Analysis and Command (PVPAC), and Flight Path Analysis and Command (FPAC) will be handled through a command coordinator before they are routed to the Space Flight Operations Director (SFOD) in Mission Control. In general, commands will be assembled in blocks and time-tagged according to the desired time of execution. Commands intended for immediate transmission will be tagged accordingly.

  Following SFOD approval, the DSS will be notified of impending command activity with Planetary Vehicle 1 or 2. The block of commands will then be sent to the SFOF computer which will affix addresses and time tags, generate block signal command messages, and display the commands to be transmitted. In addition, the computer will verify the commands against a permissive command list loaded into the computer to ensure that the command is compatible with the mission constraints for the current phase of the operation and with the remainder of the command lists in the spacecraft command programmer.

Upon notification from the DSS manager that uplink spacecraft lock-on has been achieved and the system is ready to command, the command message upon authorization from the SFOD will be transmitted from the SFOF computer via the NASCOM communications processor to the DSS communications processor where a retransmission check will be made. The command message will then be loaded serially into the DSS/TCP computer memory from the DSS communications processor.

The TCP computer will format the commands in a manner compatible with the command encoder. Upon receiving the serially entered commands from the TCP, the command encoder will transmit the message automatically to the spacecraft. During command transmission, the MIE monitor receiver will detect the PSK subcarrier frequency and send it to the computer buffer where each digital command bit will be detected and then compared with the command bit structure in the TCP computer memory. If an incorrect bit is detected, the remainder of the command transmission will be inhibited. Command logging will be maintained by the computer.

- Emergency Mode. This mode of operation will be used when either the SFOF computers, the command data transmission link to the DSS, or the DSS/TCP is unavailable. Based on direction from the Space Flight Operations Director (SFOD), commands will be manually inserted into the command encoder via digital switches. Addresses will be manually selected and the spacecraft commands will be transmitted after a permissibility check is made using the TCP computer. In this mode a separate computer program will be loaded into the station computer for permissibility checks. The process during command transmission will be the same as in the automatic mode including the monitor and inhibit functions. In the event the TCP computer is unavailable, the permissive checks will be performed by the operator monitoring visual displays.

For test purposes the command encoder will be capable of simulating the output of the monitor receiver to the computer buffer. Also, the commander encoder will be used for checkout and verification of proper operation of the command system by local station personnel.

The command encoder will monitor the permissive command and verification checks performed by the TCP computer on command messages being transmitted from the SFOF via the DSS to the spacecraft. Indications of inhibited commands and detection of message errors will be displayed on the front panel of this equipment.

3.12 VIDEO RECONSTRUCTION (MDE)

The MDE system design will provide for near-real time data processing at the SFOF and HOSC of planetary vehicle photographic and television data. Upon receiving the planetary vehicle data, the DSS will demodulate and decode the video information contained in the PCM data stream for transmission to the SFOF and HOSC via wideband data links. The video data will then be routed to the image reconstruction electronics at the SFOF and HOSC for picture reconstruction and data quality monitoring and assessment. Magnetic recordings of the video data received at the DSN and HOSC will be made for subsequent picture enhancement using the third-generation (3G) computer at the SFOF.

The image reconstruction electronics will be capable of reconstructing the video PCM data received at the following bit rates: 51.2, 25.6, 6.4, and 3.2 kb/sec. The pictures will be reconstructed on either 35- or 70-mm motion picture film, depending on the number of scan lines per frame. Processing of the film will be performed in a video processing laboratory at SFOF and the HOSC including color reproduction.

Scan conversion equipment will be provided at the SFOF to convert reconstructed video data scan frequencies to the RETMA 525 line scan format for use with the closed-circuit television displays in order to provide near-real-time video data to the user areas and commercial systems.

4. MDE OPERATIONAL TEST AND CHECKOUT EQUIPMENT

The test operational MDE hardware consists of the MDE system test set, (Figure 4-1). Standard test equipment is comprised of a circuit module tester. The MDE test equipment, except the standard circuit module tester, interconnects with the DSN MDE by the MDE test cable set. System requirements of the MDE operational test equipment are as follows.

4.1 MDE TEST EQUIPMENT

The MDE test equipment will be comprised of the following items:

- Data format generator
- Error rate tester
- Test transponder
- RF patch panel.
- Module tester.

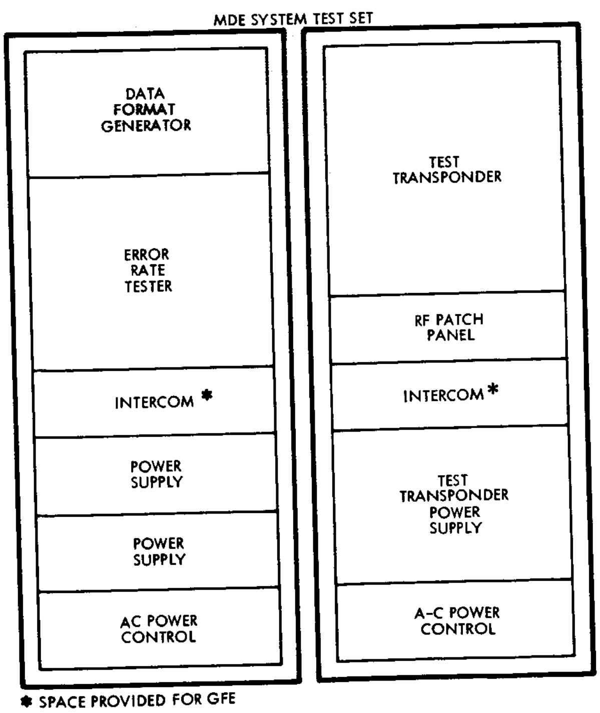


Figure 4-1
MDE SYSTEM TEST SET is interconnected with the in-line MDE at DSN via test cables to test the communication system performance and threshold levels.

4.1.1 Data Format Generator

The data format generator will be capable of generating Link 1, 2, and 3 subcarrier signals of the spacecraft, modulated with selectable serial telemetry data, simulating the normal outputs of the DSIF receiver to the demodulator/synchronizer. In addition, this unit will be capable of simultaneously generating serial biorthogonally encoded or unencoded data at any of the spacecraft transmission bit rates with all telemetry data frame constants properly located within the frame. This capability will simulate the normal demodulator/synchronizer output to the biorthogonal decoder or computer buffer. The data format generator will contain front panel switches which will control the generation of telemetry data to simulate the Voyager spacecraft telemetry system in all modes, formats, and transmission bit rates.

4.1.2 Error Rate Tester

The error rate tester will be used to provide the capability of frequency checks on the operation of the demodulator/synchronizer and also on associated data processing equipment. It is intended that the error rate tester serve as a unit-level trouble-shooting device as well as for operational readiness testing. In particular, it will enable the station operators to conduct tests of the demodulator/synchronizer performance with noisy signal conditions that closely simulate the station receiver output. The unit will be capable of generating signals representing real-time biorthogonally coded or unencoded telemetry data at any of the spacecraft bit rates mixed with white Gaussian noise at adjustable levels to simulate the DSIF receiver detected output.

The signal-to-noise ratio will be established by the operator selecting the desired level. Flexibility in choosing the length of the test (and thus its statistical accuracy) will be provided.

The input to the tester will consist of the reconstructed data from the demodulator. Provision will be made for an auxiliary test signal input such as from the data format generator. Outputs will consist of the simulated signal plus noise and a bit clock output. This latter output will be capable of being used to replace the bit synchronizing function in the demodulator under test.

4.1.3 Test Transponder

The test transponder and transponder power supply will be capable of simulating the Voyager spacecraft RF transmission to test compatibility of each DSS station with the Voyager spacecraft command and telemetry communication links.

The test transponder will perform the following general functions:

- Receive an S-band signal that is PSK phase modulated with binary coded command information

- Demodulate, decode, and display the command information, and provide a command bit stream output for recording on a strip chart recorder

- Generate S-band PSK/PM biorthogonally coded or uncoded signals with Links 1, 2, and 3 binary-coded telemetry information from the Voyager data format generator or a magnetic tape recorder

- Provide appropriate variable attenuators to reduce the received signal (and the generated signal) to a level near the threshold of the test transponder (and DSS station) receiver.

In order to provide a good simulation of the Voyager spacecraft, the transponder will utilize spacecraft components insofar as practicable.

The mechanical design of the test transponder will provide for operation in either the DSIF collimation tower room or the DSIF Voyager mission oriented room. The former location will require that the rack containing the test transponder and its power supply be portable.

4.1.4 RF Patch Panel

A test transponder patch panel will be provided in the same rack containing the test transponder for the purpose of providing input and output signal paths to and from the transponder.

4.1.5 Module Tester

The MDE circuit module tester will be a standard off-the-shelf item. The device will be manually operated in conjunction with a standard laboratory oscilloscope.

5. MDE CONFIGURATION

The Voyager MDE for the DSS will consist of six racks of equipment (Figures 3-1 and 4-1) and a bench mountable module tester. Four of the equipment racks contain in-line MDE and are designated the telemetry data processing subsystem, the biorthogonal decoder subsystem, and the command generation subsystem. These latter racks constitute in-line equipment to provide each of the Deep Space Stations with a capability for command transmission to and telemetry reception from the Voyager spacecraft. The telemetry data processor consists of two demodulator/synchronizers (one for each spacecraft) and their corresponding power supplies in one rack and two computer buffers and their power supplies in another rack. The biorthogonal decoder consists of a rack with two decoders and two power supplies. The command generator consists of a rack containing two command encoders, one of which is a spare, and two power supplies. Backup in-line MDE consists of one spare assembly (not rack-mounted) for each demodulator/synchronizer, computer buffer, and biorthogonal decoder. Since all power supplies are identical except those used with video reconstruction equipment, only one spare will be provisioned for each station.

The remaining two equipment racks contain an MDE system test set. One of these racks contains a data format generator, error rate tester, and their corresponding power supplies. The other rack contains a test transponder, RF patch panel, and test transponder power supply. Since this is not in-line equipment, no assembly-level spares are provisioned.

Voyager MDE for the SFOF and HOSC will consist of eleven racks of in-line equipment. Two of the racks contain the telemetry data processing subsystem and another the biorthogonal decoders. Three more of the racks, identified as the spacecraft subsystem display (Figure 3-3) contain computer-driven equipment which provides the SFOF and HOSC with a backlighted diagram status display of the spacecraft subsystems performance as appropriate for each subsystem. The remaining five racks (Figure 3-2), designated as the image reconstruction subsystem, contain equipment for video data picture reconstruction. This equipment also includes a unit containing optics and film.

In addition to the in-line MDE, a rack of test equipment will be provided to the SFOF and HOSC. This rack will contain a data format generator, error rate tester and their corresponding power supplies.

6. MDE HARDWARE DESCRIPTION

This section provides a technical description of each assembly or subsystem of the MDE and the functions they are designed to perform.

6.1 DEMODULATOR/SYNCHRONIZER (LINK 1)

6.1.1 Functional Description

Figure 6-1 is a block diagram of the Link 1 demodulator/ synchronizer. The unit extracts data and generates a clock signal from a NRZ-C coded PCM bit stream, modulated on a 1.024 MHz subcarrier, under conditions of very poor signal-to-noise ratio.

The unit operates in two separate computer selectable modes. In mode 1 the bit stream received from the spacecraft is biorthogonally encoded (Symbol rate = 2.73 kb/sec). In mode 2 the input bit stream is not biorthogonally encoded (symbol rate = bit rate = 512 bits/sec).

The biphase modulated subcarrier input signal in mode 2 has a very high subcarrier frequency to data ratio, $(1.024 \times 10^6/512) = 2000$ subcarrier cycles per bit. It is therefore difficult to implement an optimum detection scheme since no synchronization energy is transmitted in a biphase modulated subcarrier. The in-phase quadrature (I-Q) demodulator/ synchronizer detection scheme is near optimum only in the case where the input signal subcarrier frequency to data ratio is not very high. An alternative to the I-Q demodulator/synchronizer detection scheme for mode 2 is to use an Apollo type, nonoptimum, double and divide-by-two subcarrier demodulation technique along with the bit synchronizer section of an I-Q demodulator/synchronizer. This can be accomplished because of the predetection filtering available in the Apollo baseband separation unit which improves the signal-to-noise ratio prior to demodulation. The baseband separation unit is not employed in mode 1 because the nonoptimum detection technique would negate the advantages of biorthogonal encoding. Furthermore, the mode 1 subcarrier frequency to data (symbol rate) ratio is much lower and the near-optimum detection (I-Q demodulator/ synchronizer) technique can be employed.

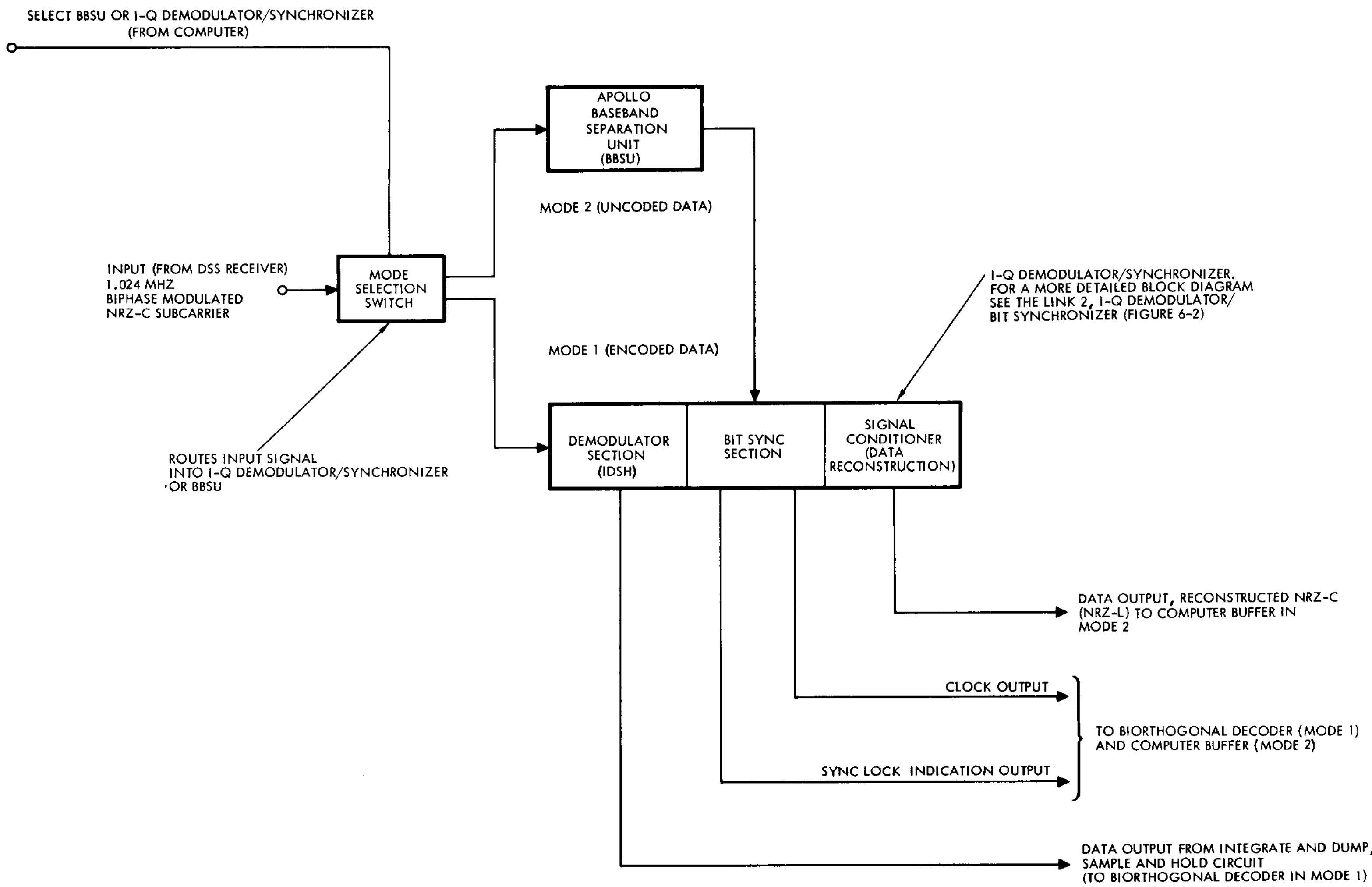


Figure 6-1
LINK 1 DEMODULATOR/SYNCHRONIZER extracts data and generates a clock signal from the NRZ-C coded PCM bit stream modulated on a 1.024 MHz subcarrier.

In mode 1 the baseband separation unit is by-passed and the input signal is routed directly into the I-Q demodulator/synchronizer via the mode selection switch. The detected data output is taken from the integrate and dump, sample and hold (I and D, S and H) circuit in the demodulator section of the I-Q demodulator/synchronizer and sent (with a clock signal) to the biorthogonal decoder for processing. The link 1 I-Q demodulator/synchronizer is similar to the Link 2 I-Q demodulator/synchronizer shown in detail in Figure 6-2.

In mode 2 the input signal is routed to the baseband separation unit via the mode select switch. The baseband separation unit removes the 1.024 MHz subcarrier and passes the detected noisy NRZ-C bit stream to the signal conditioner and bit sync section of the I-Q demodulator/synchronizer where the clock and reconstructed NRZ-C (NRZ-L) data are generated. The reconstructed NRZ-L data and clock are then sent to the computer buffer.

6.1.2 Performance Requirements

a) Design Parameters:

1)	Symbol Rate (Mode 1 biorthogonally encoded)	2.73 kb/sec
2)	Bit Rate (Mode 2 spacecraft biorthogonal encoder by-passed)	512 bits/sec
3)	Bit Error Rate	See link 2 bit error rate
4)	Acquisition Time (Mode 1)	

Narrowband*. With a SNR of 11.8 db measured in a bandwidth equal to one half the symbol rate and with a transition density of 50 percent, subcarrier lock shall occur within two sweeps of the demodulator voltage controlled oscillator. At 2.73 kb/sec the nominal narrowband period shall be 30 seconds.

*See Link 2 functional description for definition.

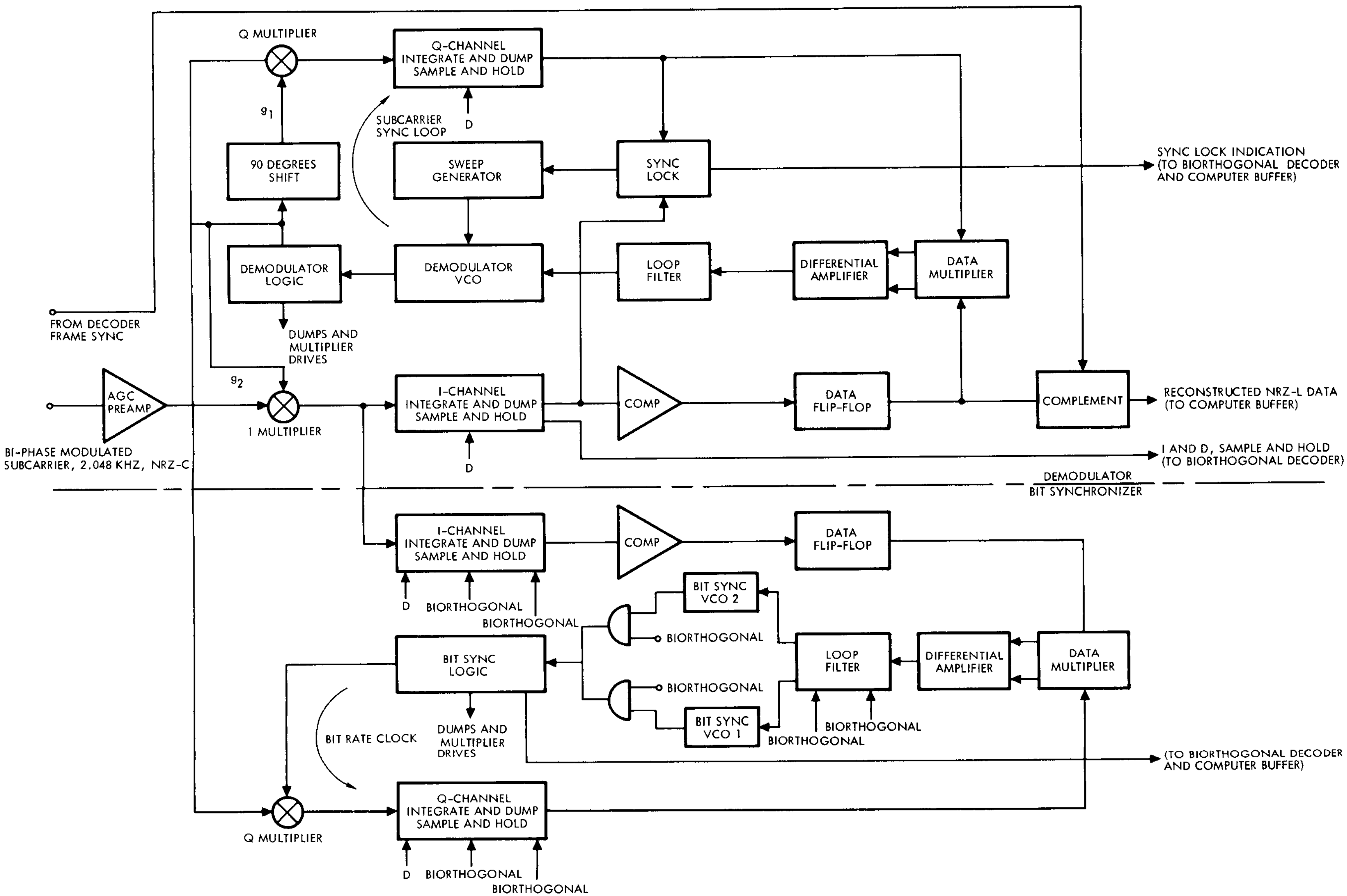


Figure 6-2
LINK 2 IQ DEMODULATOR/SYNCHRONIZER extracts data and a clock signal from a 2.048 Hz square wave subcarrier that is biphased modulated with NRZ-C data.

Wideband*. With a SNR of 18.3 db measured in a bandwidth equal to one half the bit rate and with a transition density of 50 percent, subcarrier lock shall occur within two sweeps of the demodulator VCO. At 2.73 kb/sec the nominal wideband sweep period shall be 15 seconds.

5) Acquisition Time (mode 2)

With a SNR of 18.3 db measured in a bandwidth equal to one half the bit rate and with a transition density of 50 percent subcarrier lock shall occur within 30 seconds.

6) Tracking Range	0.1 percent of the subcarrier rate
7) Capture Range	0.1 percent of the subcarrier rate
8) Minimum Transition Density to Maintain Lock	15 percent

b) <u>Signals, Input and Output</u>

1) Input Signal	Sinewave, NRZ-C noisy bit stream biphase modulated on a 1.024 MHz subcarrier.
2) Input Bit Rate Jitter	The rms bit rate jitter on the input signal shall be less than 0.01 percent of the bit period duration.
3) Subcarrier Frequency Stability	±0.05 percent
4) Frame Sync Input Signal*	Bilevel input signal from the computer buffer to indicate frame sync.
5) Output Signals	
Clock output Sync lock indication output	Sent to the biorthogonal decoder and the computer buffer.
Reconstructed NRZ-C (NRZ-L) data output (mode 2)	Sent to the computer buffer
Data output from integrate and dump, sample and hold (Mode 1)	Sent to biorthogonal decoder
VCO selected; Mode 1, Mode 2*	

*See Link 2 Functional Description for definition

c) Displays

1) Sync lock visual indication

2) VCO selected; Mode 1, Mode 2

3) Data output complemented*

d) Controls

The unit shall be capable of performing the following functions either remotely (computer control) or by means of manual controls on the front panel:

1) VCO Select; Mode 1, Mode 2

2) Power on/off

e) Interfaces

The input to the equipment shall originate from the telemetry receiver (MIE) or, under certain conditions, the input signal may originate from a magnetic tape recorder. The outputs will be supplied to the computer buffer (MDE) and the biorthogonal decoder (MDE).

f) Constraints

When the input signal to the unit is derived from the magnetic tape recorder it is expected that the resulting bit rate jitter on the input signal will increase. Therefore, in order to maintain bit error rate performance the SNR must be increased.

g) Physical Characteristics

This unit will be housed in a slide mounted drawer assembly which consists of plug-in circuit cards.

6.2 DEMODULATOR/I-Q BIT SYNCHRONIZER (LINK 2)

6.2.1 Functional Description

Figure 6-2 is a block diagram of the link 2 I-Q demodulator/synchronizer. The purpose of the unit is to extract NRZ-L data and a clock signal from a 2.048 kHz square wave subcarrier that is biphase

*See Link 2 Functional Description for definition modulated with NRZ-C data under conditions of very poor signal-to-noise ratio. The clock signal is used by the biorthogonal decoder and the computer buffer unit. The integrate and dump, sample and hold (I and D, S and H) output is used by the biorthogonal decoder when the PCM bit stream is biorthogonally encoded. The reconstructed data output is used by the computer buffer when the spacecraft biorthogonal encoder is bypassed. The demodulator/synchronizer comprises two major sections: a subcarrier demodulator section, and a bit synchronizer section. The two sections employ similar phase-lock loop circuits. The subcarrier demodulator section locks on to the phase of the incoming subcarrier, determines the sense of each received bit, and in the case of uncoded data produces a noise free data bit stream, which is sent to the computer buffer. When the input data is encoded the output of the I channel (IDSH) is sent to the biorthogonal decoder. The bit synchronizer section locks on to the bit rate of the received signal and generates a stream of bit-clock pulses coherent with the received bit rate. Each section contains an I-Q channel. The I-channel in each section consists of an I and D, S and H circuit which is driven by an I-multiplier. The I and D, S and H drives a data flip-flop via a comparator circuit. The output of the data flip-flop provides a reference for the data multiplier which is used in the Q-loop to remove sense ambiguities, due to information modulation, from the VCO correction signal. The VCO in the demodulator section is always maintained in frequency coherence with the subcarrier frequency. The operating VCO in the bit synchronizer section is always maintained in frequency coherence with the incoming bit rate. Two VCO's are used in the bit synchronizer section to accommodate the two possible true bit rates. One VCO is provided for the 42.66 bits/sec rate (biorthogonal encoded bit stream) and the other VCO is provided for the 8 bits/sec rate when the spacecraft birothogonal encoder is bypassed.

The sync indication and acquisition control functions are generated by subtracting the averaged output of the demodulator Q-channel I and D, S and H from the averaged output of the demodulator I-Channel I and D, S and H. Sync indication is obtained when the I-Q function is a maximum.

Since the NRZ-L code on the input subcarrier is nondifferential, the polarity of the input code depends upon the absolute phase between the input and the g_1, g_2 reference functions (see Figure 6-2). The frame sync signal from the computer buffer can be used to resolve automatically the ambiguity. If the unit does not receive a frame sync signal within a specified period after it has given a sync lock indication, the NRZ-L output will be automatically complemented to obtain the correct data output polarity.

The demodulator/synchronizer may be operated with either of two selectable demodulator loop bandwidths, narrowband or wideband. The normal mode is the narrowband mode. The wideband mode provides a faster tracking rate at the expense of poorer noise rejection. Two conditions under which the wideband mode is useful are when a subcarrier source has jitter (tape input), and during acquisition, when the increased bandwidth may decrease acquisition time. Switching from narrow to wide-band width involves changing several time-constants in the demodulator I and Q channels and changing the demodulator integrators from active integrate-and-dump circuits to active single pole filters.

6.2.2 Performance Requirements

a) Design Parameters

1)	Symbol Rate (Biorthogonally encoded)	42.66 bits/sec
2)	Bit Rate (Spacecraft biorthogonal encoder bypassed)	8 bits/sec

3) Bit Error Rate

Narrowband Mode: The demodulator/synchronizer shall achieve a bit error rate of 10^{-3} or less for non-return-to-zero change (NRZ-C) code with an input signal to noise ratio (SNR) of 11.8 db in a bandwidth equal to one half the bit rate. For SNR's greater than 7.3 db, error rates shall be within the limits shown in Figure 6-3.

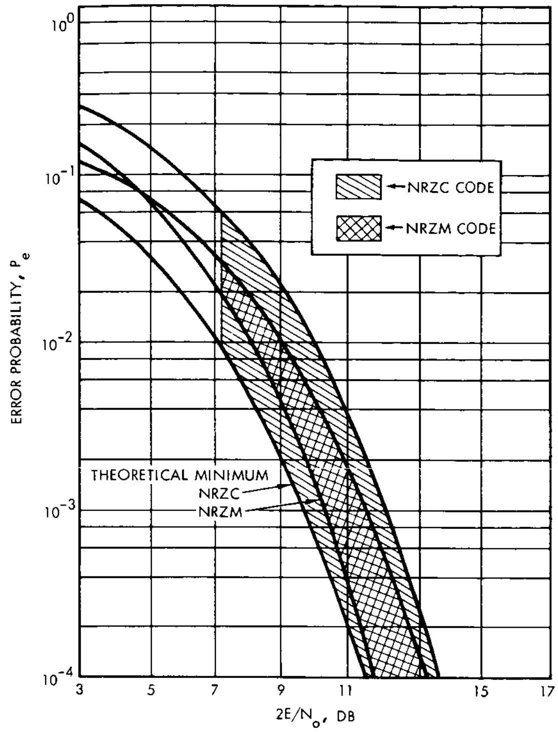


Figure 6-3
LINK 2 DEMODULATOR SYNCHRONIZER BIT ERROR probability varies with input signal-to-noise ratios in narrowband mode.

Wideband Mode: The unit shall achieve an error rate of 10^{-3} or less for NRZ-C code with an input SNR of 18.3 db in a bandwidth equal to one half the bit rate. For SNR's greater than 15.3 db, error rates shall be as shown in Figure 6-4.

4) Acquisition Time

Narrowband Mode: With a SNR of 11.8 db, measured in a bandwidth equal to one half the bit rate and with a transition density of 50 percent, subcarrier lock shall occur within two sweeps of the demodulator VCO. At 8 bits/sec the nominal narrowband sweep period shall be 320 seconds. At 42.66 bits/sec the nominal narrowband sweep period shall be 100 seconds.

Wideband Mode: With a SNR of 18.3 db measured in a bandwidth equal to one half the bit rate and with a transition density of 50 percent, subcarrier lock shall occur within two sweeps of the demodulator VCO. At 8 bits/sec, the nominal wideband sweep period shall be 80 seconds. At 42.66 bits/sec the nominal wideband sweep period shall be 45 seconds.

5) Tracking Range — 0.1 percent of the subcarrier rate

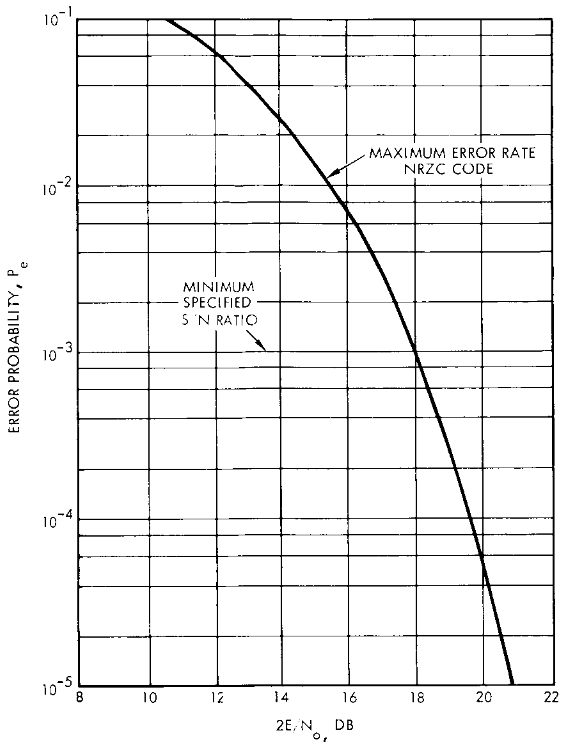


Figure 6-4
LINK 2 DEMODULATOR/SYNCHRONIZER PROBABILITY of bit error for various input signal-to-noise ratios in wide band mode.

6)	Capture Range	0.1 percent of the subcarrier rate
7)	Minimum Transition Density to Maintain Lock	15 percent

b) Signals, Input and Output

1)	Input Signal	2.048 kHz square wave subcarrier, biphase modulated with NRZ-C data
2)	Subcarrier Frequency Stability	±0.05 percent
3)	Input Bit Rate Jitter	The rms bit rate jitter on the input signal shall be less than 0.01 percent of the bit period duration.
4)	Noise Bandwidth	The input signal shall be contained in an equivalent noise bandwidth of 0 to 15 kHz single sided.

5) Frame Sync Input Signal — Bilevel input signal from the computer buffer to indicate frame sync.

6) Output Signals

Clock output	Sent to the biorthogonal decoder and the computer buffer
Sync lock indication output	
Reconstructed NRZ-L data output	Sent to the computer buffer
Integrated and dump, sample and hold output	Sent to the biorthogonal decoder
VCO Selected (42. 66 or 8 bits/sec	

c) Displays

1) Sync Lock Visual Indication

2) VCO Selected (42. 66 or 8 bits/sec)

3) Data Output Complemented

d) Controls

The unit shall be capable of performing any of the various functions tabulated below either remotely (computer control) or by means of manual controls on the front panel.

1) VCO Select (selects 42. 66 or 8 bits/sec)

2) Complement Data Output (computer buffer frame sync signal is used for remote selection of this function).

3) Power on/off

e) Interfaces

The input to the equipment shall originate from the telemetry receiver (MIE). Under certain conditions the unit input signal may originate from a magnetic tape recorder. The outputs will be supplied to the computer buffer (MDE) and the biorthogonal decoder.

f) Constraints

When the input signal to the unit is derived from the magnetic tape recorder it is expected that the resulting bit rate jitter on the input signal will increase. Therefore, in order to maintain bit error rate performance the SNR must be increased.

g) Physical Characteristics

This unit will be housed in a slide mounted drawer assembly which consists of plug-in circuit cards.

6.3 DEMODULATOR/I-Q BIT SYNCHRONIZER (LINK 3)

6.3.1 Functional Description

Figure 6-5 is a block diagram of the link 3 I-Q demodulator/bit synchronizer. The purpose of the unit is to extract NRZ-L data and a clock signal from one of six split phase change modulated subcarriers under conditions of very poor signal-to-noise ratio. The clock signal is used by the biorthogonal decoder and the computer buffer units. The I and D, S and H output is used by the biorthogonal decoder when the PCM bit stream is biorthogonally encoded. The reconstructed data output is used by the computer buffer when the spacecraft biorthogonal encoder is bypassed. The basic principles employed for demodulation and synchronization of data contained on a split phase modulated subcarrier are very similar to the principles applied in the link 2 demodulator/synchronizer. The split phase demodulator synchronizer requires only an I-Q loop to perform the demodulation and synchronization functions. The reconstructed data signal is taken directly from the data flip-flop in the I-channel. The clock output is derived from the bit rate signal generated by the bit rate counter. Again, two VCO's are required in order to accommodate the six high bit rates corresponding to the biorthogonally encoded bit stream and the six low bit rates which result when the spacecraft biorthogonal encoder is bypassed. Although the unit will process one of six basic bit rates, provision for accommodation of only two bit rates is illustrated in Figure 6-5 for simplicity.

Like the NRZ-C code on a subcarrier, split phase is nondifferential, and as a result the unit may erroneously lock on to the mid-bit transitions of the received data stream. The unit, therefore, must have an input which will bring about a correct lock-on condition. The frame sync signal from the computer buffer again can be used to automatically resolve this ambiguity. In this case, after the unit gives a sync lock indication, if a frame

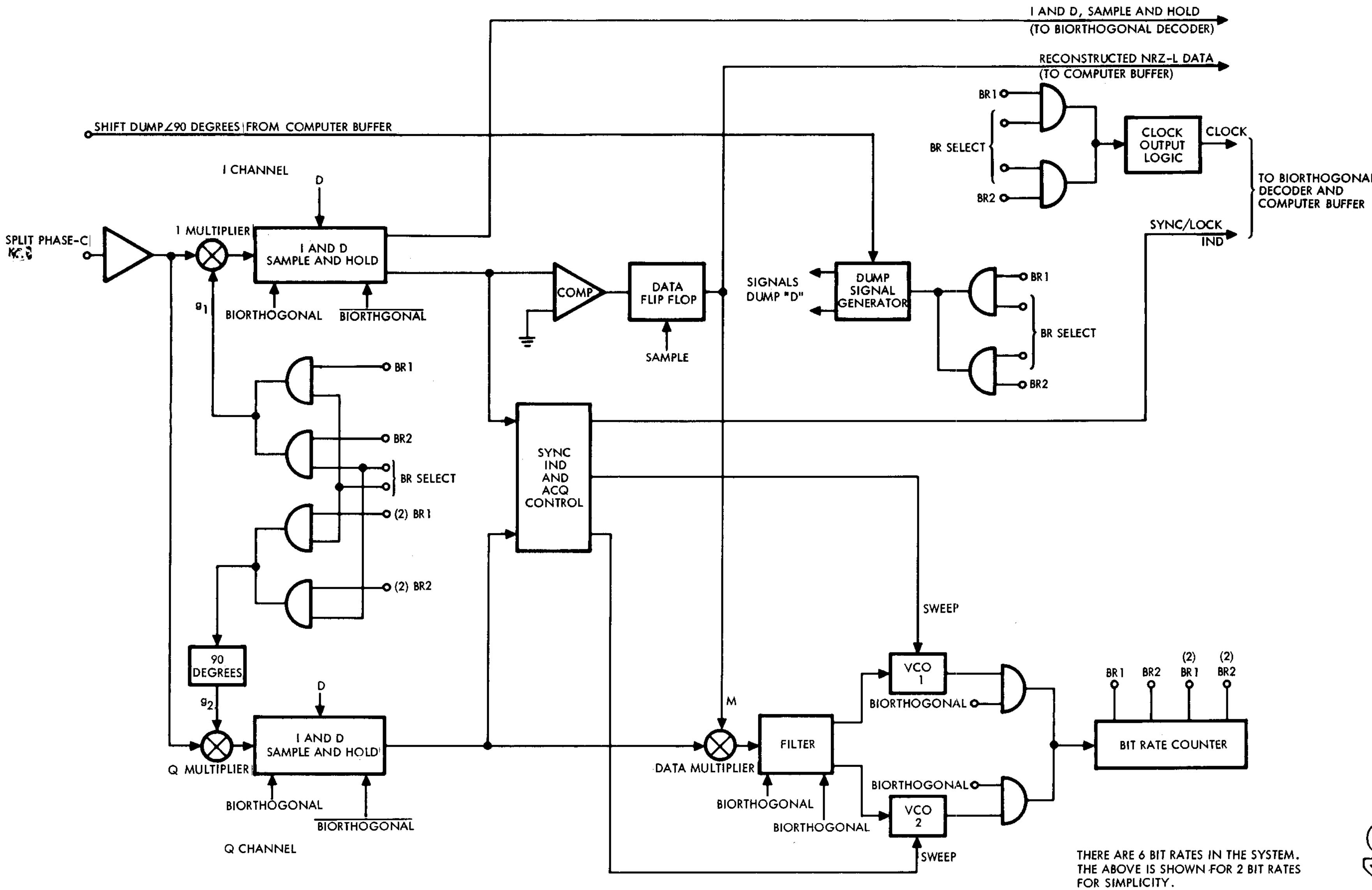


Figure 6-5
LINK 3 DEMODULATOR/BIT SYNCHRONIZER extracts data and a clock signal from one of six split phase change modulated subcarriers.

sync signal is not received from the decoder or buffer within a specified period of time the dump pulse phase will be shifted by 90 degrees to bring about a correct lock-on condition.

An alternative method that is somewhat more attractive involves sensing signal transitions at the mid-bit period. Since 100 percent transitions are guaranteed, this condition can be sensed as an indication of true bit sync. False bit sync would be accompanied by a mid-bit transition density of nearly 50 percent.

6.3.2 Performance Requirements

a) Design Parameters

1) Symbol Rates (Biorthogonally encoded)	(kb/sec) 273 136.5 68.25 34.125 17.063 2.73
2) Bit Rates (Spacecraft biorthogonal encoder bypassed)	(kb/sec) 51.2 25.6 12.8 6.4 3.2

3) Bit Error Rate

The demodulator/synchronizer shall achieve a BER of 10^{-3} or less with an input SNR of 11.8 db in a bandwidth equal to one half the bit rate. For SNR's greater than 7.3 db, error rates shall be within the limits shown in Figure 6-3.

4) Acquisition Time

With a SNR of 11.8 db measured in a bandwidth equal to one half the bit rate, signal lock shall occur within two sweeps of the VCO. The nominal sweep period for the VCO at each bit rate shall be 7 seconds.

5) Tracking Range	0.1 percent of the bit rate
6) Capture Range	0.1 percent of the bit rate

b) Signals, Input and Output

1) Input Signal (Biorthogonally encoded)

Split phase change modulated squarewave subcarrier of the following frequencies:

kHz
273
136.5
68.25
34.125
17.063
2.73

2) Input Signal (Spacecraft Biorthogonal encoder bypassed)

Split phase change modulated square wave subcarrier of the following frequencies:

kHz
51.2
25.6
12.8
6.4
3.2

3) Input Bit Rate Jitter

The RMS bit rate jitter on the input signal shall be less than 1 percent of the bit period duration.

4) Frame Sync Input Signal

Bilevel input signal from the computer buffer to indicate frame sync.

5) Output Signals

Clock output

Sync Lock Indication output

Sent to the biorthogonal decoder and the computer buffer

Reconstructed NRZ-L data output

Sent to the computer buffer

Integrate and dump, sample and hold output

Sent to the biorthogonal decoder

VCO selected

c) Displays

1) Sync Lock Visual Indication

2) VCO Selected

3) Bit Rate Selected

4) Dump Shifted 90 degrees

d) Controls

The unit shall be capable of performing any of the various functions tabulated below either remotely (computer control) or by means of manual controls on the front panel.

1) Bit Rate Select (6)

2) Shift dump 90 degrees (incorrect bit sync signal, from the computer buffer, is used for selection of the function).

3) VCO select (selects a VCO frequency corresponding to the high or low frequency bit rates).

4) Power on/off.

e) Interfaces

The input to the equipment shall originate from the telemetry receiver (MIE). Under certain conditions the unit input signal may originate from a magnetic tape recorder. The outputs will be supplied to the computer buffer (MDE) and the biorthogonal decoder (MDE).

f) Constraints

When the input signal to the unit is derived from the magnetic tape recorder it is expected that the resulting bit rate jitter on the input signal will increase. Therefore, in order to maintain bit error rate performance the SNR must be increased.

g) Physical Characteristics

This unit will be housed in a slide mounted drawer assembly which consists of plug-in circuit cards.

6.4 BIORTHOGONAL DECODER

The biorthogonal decoder associates transmitted 32 symbol word groups with unique 6-bit information words in accordance with a pre-assigned tabular relationship. There are 32 possible 32-bit codes and their complements (64 total 32-bit words) to be compared with each 32-bit symbol word group that is received. The comparison is accomplished by digital correlation techniques in real time so that the decoder outputs NRZ-C data consisting of 6-bit information words.

The decoder receives a clock signal and the integrated NRZ-C data plus noise from the demodulator/synchronizer associated with Link 1, 2, or 3. The decoded output information is sent to the computer buffer for signal conditioning before entering the computer.

The primary internal functions of the decoder are:

- To digitize the detected serial information symbols
- To sequentially generate a code dictionary on a cyclic basis for real time correlation
- To sequentially correlate each received code word with the dictionary codes (in real-time)
- To detect the most likely data word received based upon the largest generated correlation magnitude
- To acquire word synchronization
- To provide signals for computer measurement of received SNR

The mechanization of the decoder is shown in simplified form in the block diagram of Figure 6-6. The integrated NRZ-C data plus noise and a clock (symbol sync) signal is selected (manually or by computer control) from the demodulator/synchronizer in Link 1, 2, or 3. The data signal is sampled, held and digitized by the analog-to-digital converter. The digitized value of each symbol in a symbol word group is sent to a Mod 2 adder where it is combined (Mod 2) with a comma-free vector for word synchronization purposes (to be explained later). When the decoder has acquired word synchronization the Mod-2 adder output consists of the original biorthogonal code. The digitized serial symbol word group is shifted into the digital correlator where it is sequentially correlated with each of the 32 dictionary words generated by the biorthogonal code generation logic. The output of the digital correlator will consist of a serial bit stream that represents the correlated values of the received

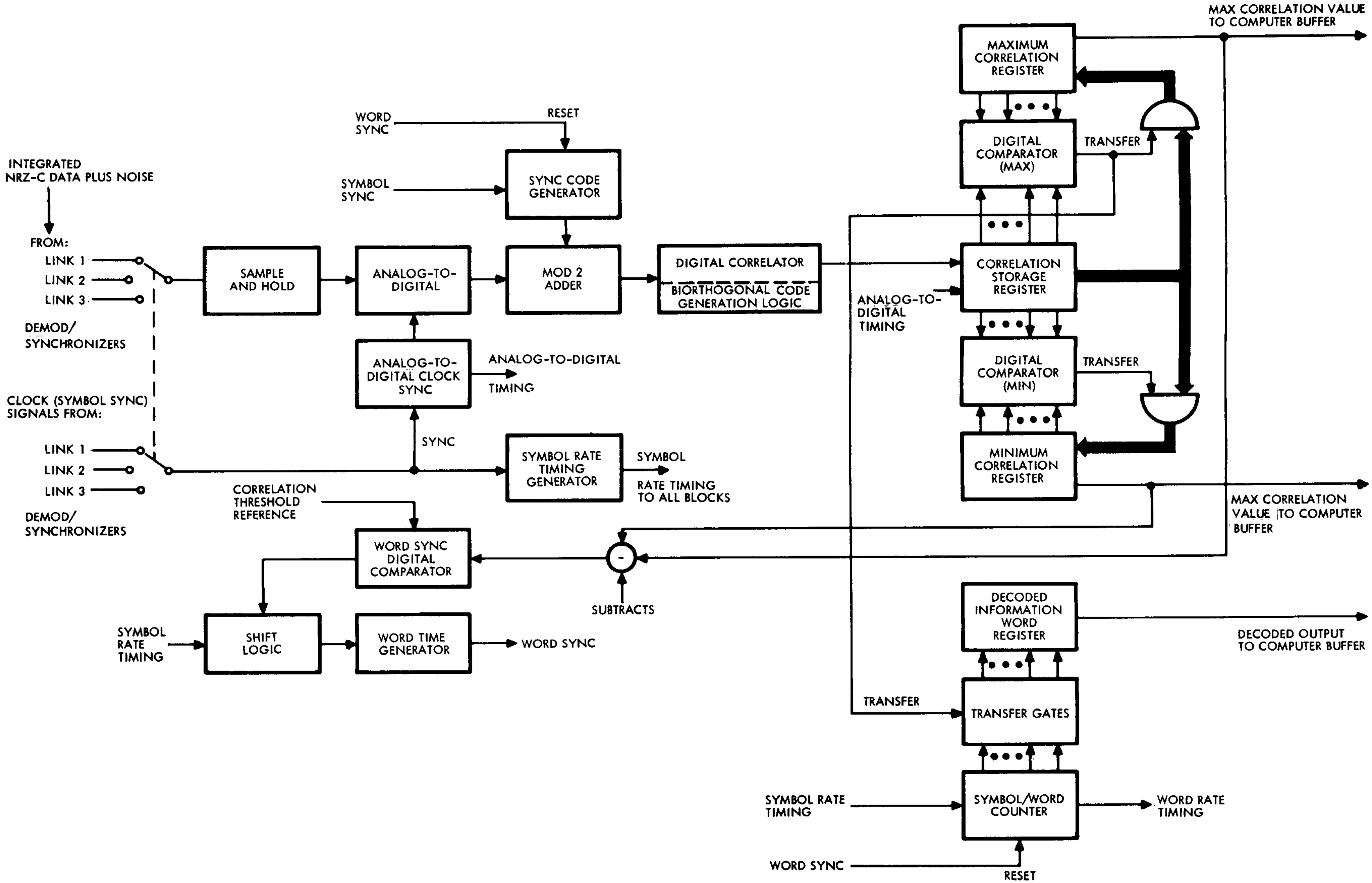


Figure 6-6
BIORTHOGONAL DECODER associates transmitted 32 symbol word groups with unique 6-bit information words in accordance with prescribed tabular relationships.

input symbol word group with the 32 dictionary code words. Thirty-two correlation values in digital form will appear at the output of the correlator for each symbol word group received. Since the correlation technique generates a sign for each correlation value, it is unnecessary to compare the received symbol word group with the 64 possible dictionary code words. Therefore only 32 correlations need be generated because a negative correlation value will correspond to having received the complement of the orthogonal code word in the dictionary that produced the given correlation magnitude.

The 32 correlation values from the digital correlator for each received symbol word group are serially shifted into the correlation storage register where they are sequentially compared with the maximum and the minimum smallest yet (magnitude comparison) received correlation values for the received word group. The largest and smallest yet comparisons are performed simultaneously in parallel fashion. When a smaller or larger correlation magnitude is detected by a digital comparator, the new value is transferred (digital comparator transfer signal) into the maximum or minimum correlation register for storage. Each time a new maximum correlation value is detected the maximum comparator transfer signal also enables the transfer gates. Enabling the transfer gates allows loading of a 6-bit information word into the decoded information word register. The 6-bit information word is the decoded value of the largest received correlation. The 6-bit information word is generated by the symbol/word counter which is updated in synchronism with the appearance of each digital correlator value.

During the processing of the correlation values for a symbol word group, the decoded information word register is continually updated with the possible decoded output value. When all correlation values have been compared the maximum correlation register will have the sign and largest correlation magnitude corresponding to the decoded 6-bit information word. The decoded information word register will contain the 6-bit information word. If the maximum correlation has a positive sign the contents of the decoded information word register are sent out. If the sign is negative the contents of the register are complemented and sent out.

Word synchronization is implemented as follows: For each received symbol word group the minimum correlation value is subtracted from the maximum correlation value in the subtractor circuit. The word sync digital comparator compares the resulting difference signal with a correlation threshold reference. Nothing is done if the subtractor difference signal is larger than the correlation threshold reference. If it is smaller, the word sync digital comparator sends a signal to the shift logic. The shift logic output will act on the word time generator. The word time generator generates a word sync signal that causes the symbol/word counter to slip sync by one symbol period. The effect of Mod 2 addition of the comma-free vector to the digitized input for synchronization can now be seen. As the process continues a point is reached where the difference between the maximum and minimum correlation values is maximized. When this difference is maximized for a given SNR, word sync has been obtained and the subtractor difference signal exceeds the correlation threshold reference. The addition of the comma-free vector to the input signal has the effect of minimizing the difference between the maximum and minimum correlation values of a received symbol word group when the decoder is not in word sync.

In addition to decoded output information, the unit also sends the maximum and minimum correlation values for a received symbol word group to the computer via the computer buffer. This information is used by the computer to determine the input SNR.

6.5 IMAGE RECONSTRUCTION SUBSYSTEM

The image reconstruction subsystem accepts digital video data, converts the data to composite video signals containing video image information and auxiliary engineering data, electronically processes the video portion of the signal and then converts the video information into a display that can be photographically recorded either on 35 or 70 mm film.

6.5.1 Functional Description

Figure 6-7 describes the system used for the image reconstruction subsystem. Pertinent parameters of the spacecraft proposed systems are

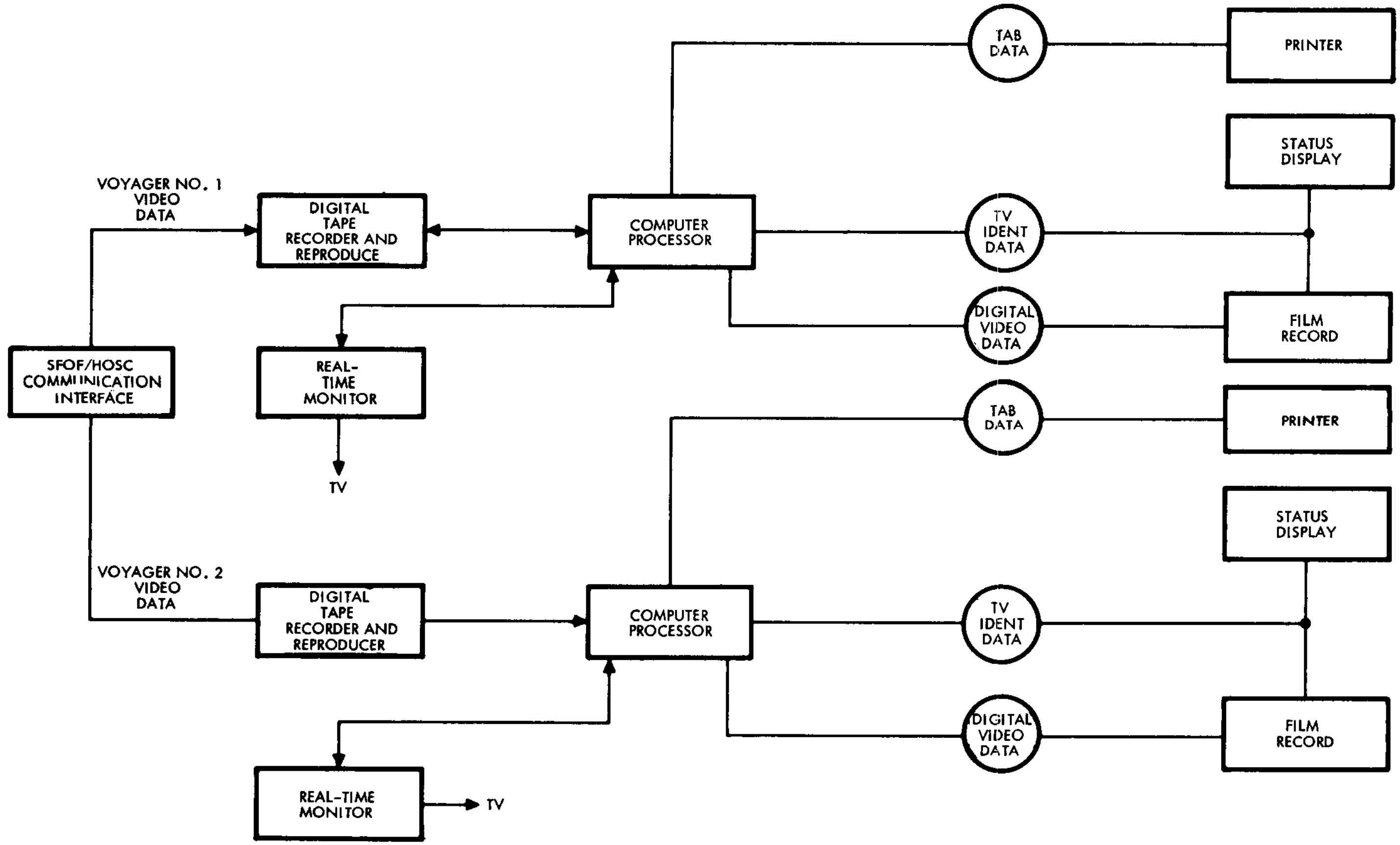


Figure 6-7
IMAGE RECONSTRUCTION SUBSYSTEM receives the digital video from the SFOF/HOSC communication center and processes it for near-real-time display and photographic recording.

tabulated in Table 6-1. The digital video data accepted from the SFOF/ HOSC communication interface is recorded in a computer-compatible format on master and duplicate tapes. The digital video data in a computer compatible format on the duplicate tapes are fed to the SFOF or HOSC computer. The computer strips out the engineering data and converts the data into a human readable format (letters and numbers). The digital video data is decoded and reformatted in the computer and is presented to the film record unit which reconstructs the data individually for each picture element. The computer also provides the capability of producing a tab data output, listing all TV elements and image data that is converted to film to provide a library of test data to support analysis of instrument performance during test and flight phases. The film record unit is a data storage device using photographic film as a storage medium. The photographic film is subsequently processed for printing in a film processing unit. The photographic film is then available for further enhancement by utilizing GFE on-site video film converters.

Table 6-1. Spacecraft Photo-Imaging Parameters

System	Type	Raster Format (mm)	Frame Scan Time (sec)	Frame Erase Time (sec)	Scan Lines	Bit Frame	Readout Time (hr at 50 kb/sec)	Ground Resolution (meters)	Data Rate Per Frame	Remarks
Hypothetical	3 x Medium Resolution	25 x 25	110.5	26	4,280	77.04×10^6	11.6	100	0.697×10^6	Return beam vidicon
Hypothetical	1 x High Resolution	25 x 25	50	20	2,860	24.33×10^6	4.8	10	0.686×10^6	Sec Vidicon
	1 x Low Resolution	18.5 x 18.5	12	20	1,430	8.58×10^6	1.56	1000	0.715×10^6	Wide angle Vidicon color
	1 x Low Resolution	18.5 x 18.5	25	26	1,400	8.35×10^6	1.56	1000	0.343×10^6	Wide angle Vidicon color
Recommended	1 x High Resolution	60 x 60			17,160	1.24×10^9	6.9	10		Film camera
	1 x Medium Resolution	60 x 60			17,160	1.24×10^9	6.9	100		Film camera
	1 x Low Resolution	18.5 x 18.5	25	26	1,430	8.35×10^6	1.56	1000	0.343×10^6	Wide angle Vidicon color
Alternate	1 x Medium Resolution	18.4 x 18.4			5,260	1.16×10^8	18.5	100		Dielectric tape framing mode
		18.4 x 100			28,500	6.3×10^8	112	100		Dielectric tape panoramic mode

The real-time monitor accepts digital video data from the computer for real-time display in the form of a picture raster. The real-time monitor provides a scan converter capability to provide RETMA television signals.

6.5.2 Equipment

6.5.2.1 Digital Tape Recorder and Reproducer Unit

The digital tape recorder and reproducer provides the means to record, store and reproduce the digital video data received at the SFOF/

HOSC complexes. The digital video data is recorded on magnetic tape in a computer compatible format.

6.5.2.2 Film Record Unit

The film record unit (Figure 6-8) is a data storage device using photographic film as a storage medium. The video data obtained by means of a telemetry link with Voyager is accurately recorded in the film record unit in the form of an analog signal on film. The unit has a bandwidth of 1.0 megacycle and both analog and digital sweeps are utilized. The equipment will operate as an on-line system with a digital data processor.

The film record unit converts the digital signal to an optical image. This is accomplished by exposing photographic film while scanning parallel horizontal lines with a flying spot scanner. The film is subsequently processed in other equipment separate from the record unit. The flying spot scanner cathode ray tube used in the film recording unit serves as a controllable light source. The light output will vary due to variations in the high voltage power supply, variations in bias voltage, and imperfections in the phosphor. These effects are removed by analog computation. A flying spot scanner monitor channel consisting of optics, photomultiplier, and electronic circuits is used to remove the undesirable cathod ray tube variations. A dynamic focus computer continuously keeps the spot in focus, regardless of its position on the flying spot scanner cathode ray tube. Spot brightness control is used to ensure that an input video signal is linearly recorded on photographic film. This is accomplished by a gamma correction network.

Data conversion is accomplished by an electron beam scanning system which utilizes both horizontal and vertical beam deflection. The scanning rate, resolution, and raster size can be varied electronically and the raster can be rotated through 360 degrees. Two image sizes, 35 and 70 mm, are employed using precision film transports. The two image sizes are generated electronically, without lens substitution.

Electronics System. The electronic system of the film record unit (Figure 6-9) provides the circuitry to perform synchronization and multiplexing, video conditioning, spot brightness control, horizontal and

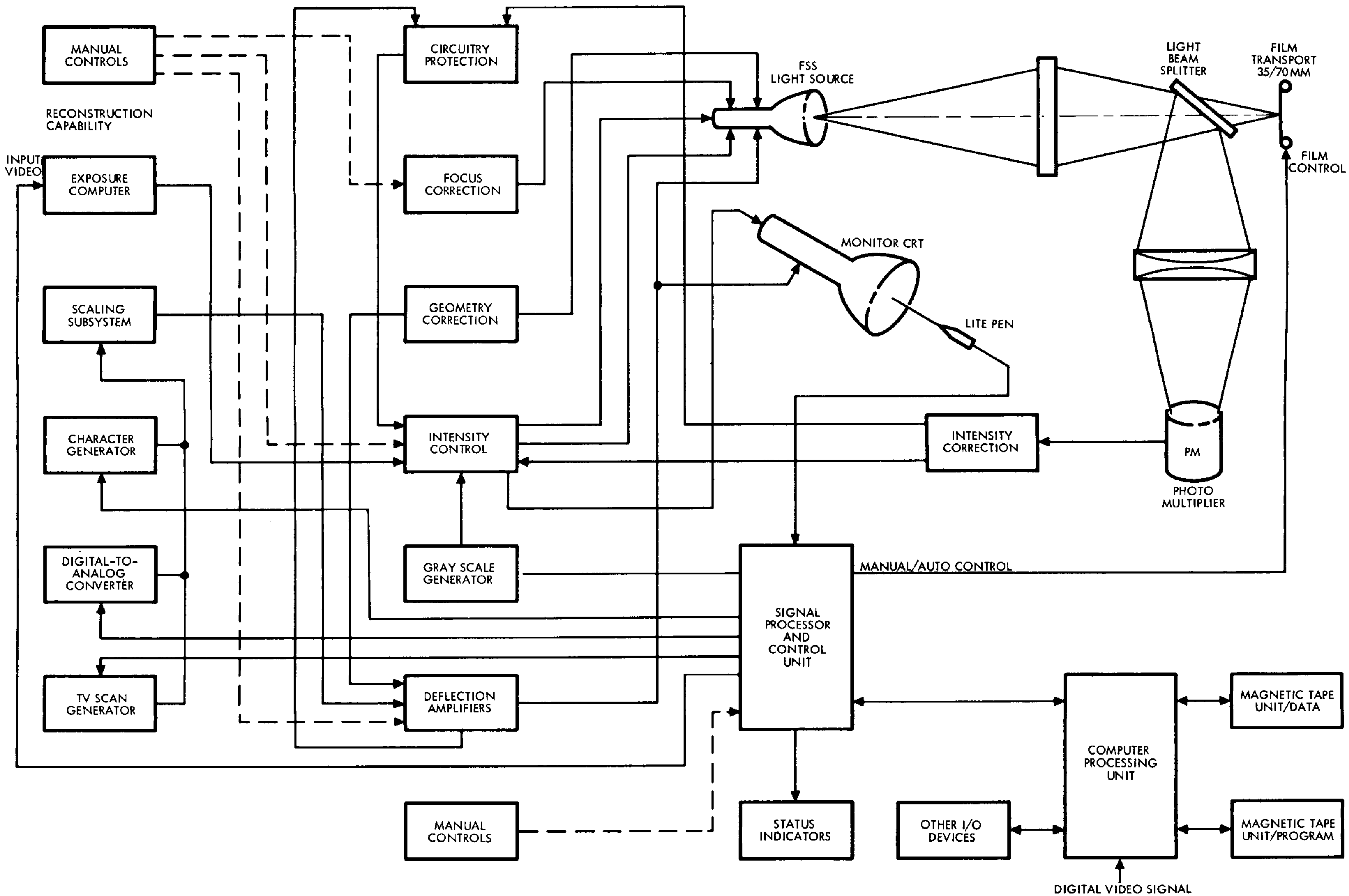


Figure 6-8
FILM RECORD UNIT receives the video data from the computer processor in its signal format, and reconstructs the picture image as photographic film.

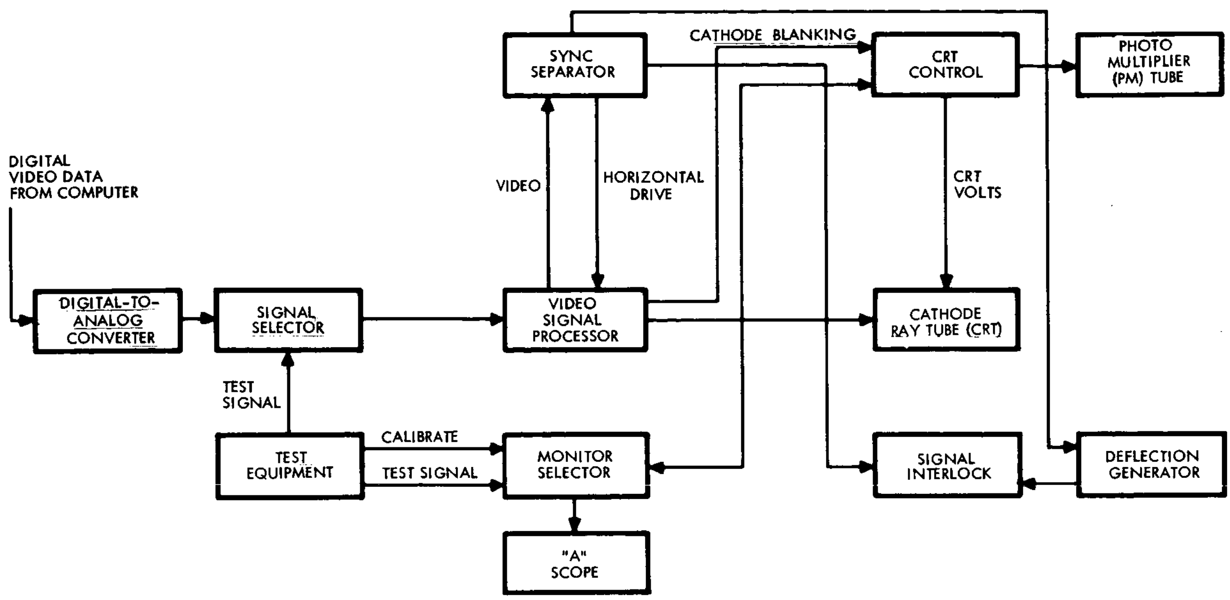


Figure 6-9
ELECTRONICS OF FILM RECORD UNIT shows the signal flow between the components and integration of the test and calibration signals.

vertical sweeps, continual focus, character generation, and converts digital video or character information to gray level information.

In order to obtain a true reproduction of the information contained in the video signal, DC response is a necessary requirement of the video amplifier. This requirement is met with the addition of an AC clamp mode. This provides the best safeguard against dc drift in the incoming signal and spurious transients. Such incoming signal variations would be seen in the recorded image as density variations and could produce false information. The ac/clamp circuit allows a droop characteristic of less than 1 percent.

To eliminate spurious signal amplitude variations above the specified white level and below the specified black level, white and black soft clipping circuits are included in the video signal processor. These circuits have adjustable threshold levels beyond which the signal is compressed so that nominally only 1/5 the signal beyond the black and white threshold levels are allowed to reach the signal grid of the cathode ray tube (CRT). Horizontal aperture equalization provides high-frequency boost and may be described as a cosine amplitude versus frequency function with linear phase characteristics.

Because of the amplitude of the video signal input and the slow scan rate, little amplification is needed to drive the CRT to obtain the necessary light output for photography. CRT drive depends upon individual tube characteristics and upon how long a particular line position has been used, but generally a gain of one or less has been found adequate in the video amplifier.

In order to display information on a CRT, synchronizing signals must be generated so that the CRT sweep will be driven in synchronism with the incoming video information. The sync pulse contained in the composite video signal therefore is stripped using a sync separator and used to phase lock a free-running oscillator. The oscillator output called the horizontal drive pulse is the timing pulse and all synchronization is derived from this output.

It is required to make the sync separator as insensitive as possible to noise. One effect of noise is to produce jitter or a time variation between the output pulses of the timing oscillator. In order for the sync separator to distinguish between the blanking pulse overshoot and the true sync pulse, a noise gate is necessary which inhibits the sync separator for that period of time from the beginning of the blanking pulse to a point 5 microseconds before the beginning of the sync pulse. The gate is enabled only after acquisition of the sync pulse, however, and prevents false sync pulses such as overshoot or ringing from influencing the time oscillator frequency. The overshoot can affect acquisition of sync only if there is excessive synchronous noise along with the overshoot on the blanking pulse. The noise gate enabling circuit, called the integrating gate, turns on the noise gate only after a signal having adequate energy persists for a relatively long time. If at any time the incoming video or the sync pulse is lost for a period exceeding 750 microseconds, the noise gate is automatically removed and the integrating gate takes over until the sync pulse is restored. The constraints imposed upon the design of the deflection generator shown in Figure 6-9 are those which will effect the geometry of the recorded image. The circuits in the deflection generator are designed to meet the following requirements:

- Linear sweep
- Minimum drift in vertical and horizontal line position and length
- Minimum line curvature regardless of position on face of tube
- Maximum correction for flatface distortion
- Dynamic focus
- Line positioning control
- Option of horizontal or raster scan
- Vertical spot wobble
- Focus symmetry
- Sweep loss protection.

The horizontal drive pulse output from the sync separator provides synchronization of pulses fed to the deflection generator which in turn initiates each of the horizontal scan lines.

The deflection circuitry utilizes current feedback in each of the vertical and horizontal drivers to insure that the current is truly linear with respect to the generating voltage. The sweep-generating circuits include an analog precision ramp generator adjustable from 2 to 2000 lines/sec for the horizontal drive, an alternative digital horizontal stepping circuit allowing from 100 to 4000 discrete positions, and a digital circuit adjustable from 100 to 4000 lines/frame for the vertical deflection. The purely digital vertical deflection is used in view of the extremely long frame times anticipated that would require some extreme component values in an analog circuit. In addition, either the horizontal or the vertical circuits can be driven from external signals to generate either entirely independent sweeps or minor sweep deflections.

The light path contains a beam splitter (Figure 6-8) which diverts part of the light into a photomultiplier tube. This is a feedback input to an operational amplifier which closes a servo loop around the CRT and includes the light path, insuring that the light is a reasonably linear function of the input voltage. The linearity depends on how much loop gain can be used, with the normal servo problem of attaining a sufficiently high loop gain while keeping the loop stable. This in turn necessitates the use of a phosphor of the P-16 type which decays to less than 10 percent response after 5 microseconds. With this scheme the loop can be kept closed past 1 megacycle. With the P-16 phosphor and with a brightness on the face of the tube which will give a density on film of about 1.3 when scanning 2.5^2 cm/sec in a uniform raster with film processed to gamma = 1, a half-amplitude spot diameter of about 30 microns can be obtained on the film without dynamic focus correction over a 1 x 1 inch raster.

Scanning spots have an intensity distribution approaching the Gaussian, thus preventing the production of extremely sharp edges. Specifically, the scanning spot acts as a low pass filter, with no transmission above a spatial frequency (cycles/mm) at which the diameter of the spot is equal to or greater than the period. If the spot diameter can be kept small compared to the size of the information to be recorded, this loss of high frequency data can be avoided. Thus, if the data contains an upper frequency of 20 cycles/mm the resultant spot size should be 0.001 inch diameter to give 64 percent response at this frequency. If the spot is reduced in size in an effort to get better high-frequency resolution, line-to-line ripple begins to show which may be more detrimental to the viewer than the loss of resolution caused by the larger spot. Thus the number of lines in the picture influences the useful horizontal resolution. In the vertical direction the sampling is essentially digital because of the line structure. Since the raster structure disappears when the half-amplitude diameter is approximately equal to the line spacing, this condition is generally selected as the operating point. If the line structure is considered as an attempt to reconstruct vertical frequencies in accordance with the sampling theorem, one cycle must occupy two lines so that in this direction the digital recording is at diameter/period of 1/2, giving an average response, depending on phasing, of 64 percent at this vertical frequency.

Optical System. The optical system is comprised of a flat-faced cathode ray tube (CRT), one objective lense, one beam-splitting mirror, and one photomultiplier tube. All components are mounted on a dimensionally stable material to maintain required alignment accuracies. An objective lens and a beam-splitting mirror are used to image the CRT spot onto the film. The amount of radiant flux which passes to the film is determined by the electron beam intensity, which is controlled by the video signal for that particular point. A closed-loop circuit incorporating a photomultiplier tube corrects for phosphor aging and imperfections. Quick-look capability is provided on a display oscilloscope. The oscilloscope permits portrayal of the exact same image as generated on the CRT faceplate.

Mechanical System. The system is packaged in a three bay electronics cabinet and an optics box. Approximate measurements of the three-bay cabinet are 5 feet wide, 2 feet deep, and 6 feet high. Approximate measurements of the optical cabinet are: 6 feet wide, 3 feet deep, and 5 feet high. All electronic circuitry, with the exception of the emitter follower used with the CRT brightness monitor photomultiplier, is contained on printed circuit cards designed for placement in card cages. A filtered blower provides adequate cooling. A photo-sonics transport is provided for recording and reproducing on 35 mm film. This transport uses a reversible Geneva movement to permit film movement in forward or reverse directions. The 70 mm transport also provides for bidirectional movement.

Specifications

Electronic Distortion:	$<\pm 0.1$ percent with pincushion correction
Pincushion Correction:	Electronic, analog computation
Sweep Linearity:	Better than 0.1 percent
Dynamic Focus:	Electronic, analog computation, spot focus maintained over 100 sq cm with less than 30 percent degradation at the edge of CRT
Optical Distortion:	$<\pm 0.02$ percent
System Resolution:	Spatial frequency >10 cycle/mm Shrinking raster >25 cycle/mm

Optical Spectral Response:	Compensated for P-16 phosphor
Format Rotation:	360 ±0. 1 degrees panel selectable
Format:	Image and human readable (symbol); all portions of format front panel digital switch controlled except symbols
Sizing Control:	Symbol sizing computer controlled; front panel digital switch controls image size (vertical and horizontal)
Vertical Size:	Vertical size controlled by number of lines and microns/line. 100 mm maximum, 4096 maximum number of lines
Horizontal Size and Timing:	Analog: time-350 to 500, 000 usec size - 10 to 100 mm maximum Digital: horizontal size controlled by number of elements and Microns/ element, 100 mm maximum, 4096 maximum number of elements
Offsets:	image X-axis 0 to 100 mm (right) image Y-axis 0 to 100 mm (down) symbol horizontal 0 to 100 mm symbol vertical 0 to 100 mm
CRT Intensity Control:	Instantaneous light flux monitored by PM and fed back in 1 megacycle gain bandwidth (gb) product. Feedback system utilized P-16 phosphor - CRT gamma corrected by non-linear amplifier to maintain gb product regardless of operating point
CRT Type:	Ferranti 5/71
Exposure Computer:	Linear exposure
Quick-Look Capability:	Display oscilloscope
Camera Output:	35 or 70 mm type I perforations (mechanical selection by mirror movement)
Camera Control:	Automatic and manual.

Modes:	Digital: (internal or computer controlled) random sequential search mode (TV rates for readout scanning)
CRT Protection Functions:	CRT overbright, vertical and horizontal sweep loss, maximum grid drive limiting, instantaneous blanking and protection, defocus upon sensing malfunction, power supply interlocked.
60 Cycle Noise	<1 part in 40, 000 vertical sweep

6.5.2.3 Real Time Monitor

The real time monitor provides the means to: 1) display a waveform reconstructed from the digital video data in real-time, 2) accept and identify operational commands (inserted at the console) and introduce them to the SFOF or HOSC computer. These commands will indicate significant aspects, such as listing and plotting of data after decommutation, 3) provide a magnetic disc type scan converter which accepts the digital video data from the general purpose computer and converts it to the standard RETMA 525 line television signal for use on SFOF/HOSC closed-circuit television and the public television broadcast stations.

6.5.2.4 Film Processing Unit

The film processing unit provides the means to process the 35 and 70 mm film. The processor uses a standard photographic wet immersion type machine under stringent scientific conditions. The processing chemicals are Kodak liquid X-ray developer, Kodak Rapid Fixer, and Kodak hypo cleaning agent. Processing takes place at 100°F at 7-1/2 ft/min. Both developer and printer are replenished at a rate of 25 cc/min during processing. Sensitometers and densitometers are used in conjunction with the processor to maintain rigid process control levels. Quality evaluation viewer is provided to inspect the 35 and 70 mm film.

The viewer is equipped with a stereo variable power microscope. In order to process the film for color reproduction the developed monochrome film of the spacecraft color video data is divided into three separation negatives. These three negatives are assembled to expose to a matrix film. The film is then processed through a three bath dye transfer operation to obtain a color reconstruction of the original spacecraft pictures.

6.6 COMMAND ENCODER

6.6.1 Functional Description

The primary function of the command encoder is to generate planetary vehicle command signals of three types in a form suitable for modulating the DSS command transmitter (MIE) frequency. Normal operation of the command encoder is under the control of the TCP computer (MIE) via an MDE computer buffer. As a backup capability under emergency operational conditions, the command encoder provides a manual capability for the initiation of all three types of spacecraft command signals. In addition, the command encoder may also be used to verify the operational integrity of the DSS command transmission link.

6.6.2 Design Implementation

A simplified functional diagram (Figure 6-10) illustrates the basic design composition of the command encoder. The block diagram is broken into three major sections:

- Subcarrier generator and modulation section
- Command storage and display section
- Logic control section.

The function of the subcarrier generator and modulation section is to generate the command subcarrier, provide for biphase modulating this subcarrier with the serial command data, and allow for amplitude adjustment of the composite command signal output. The subcarrier signal is derived from a tuning fork oscillator, which is a plug-in commercial

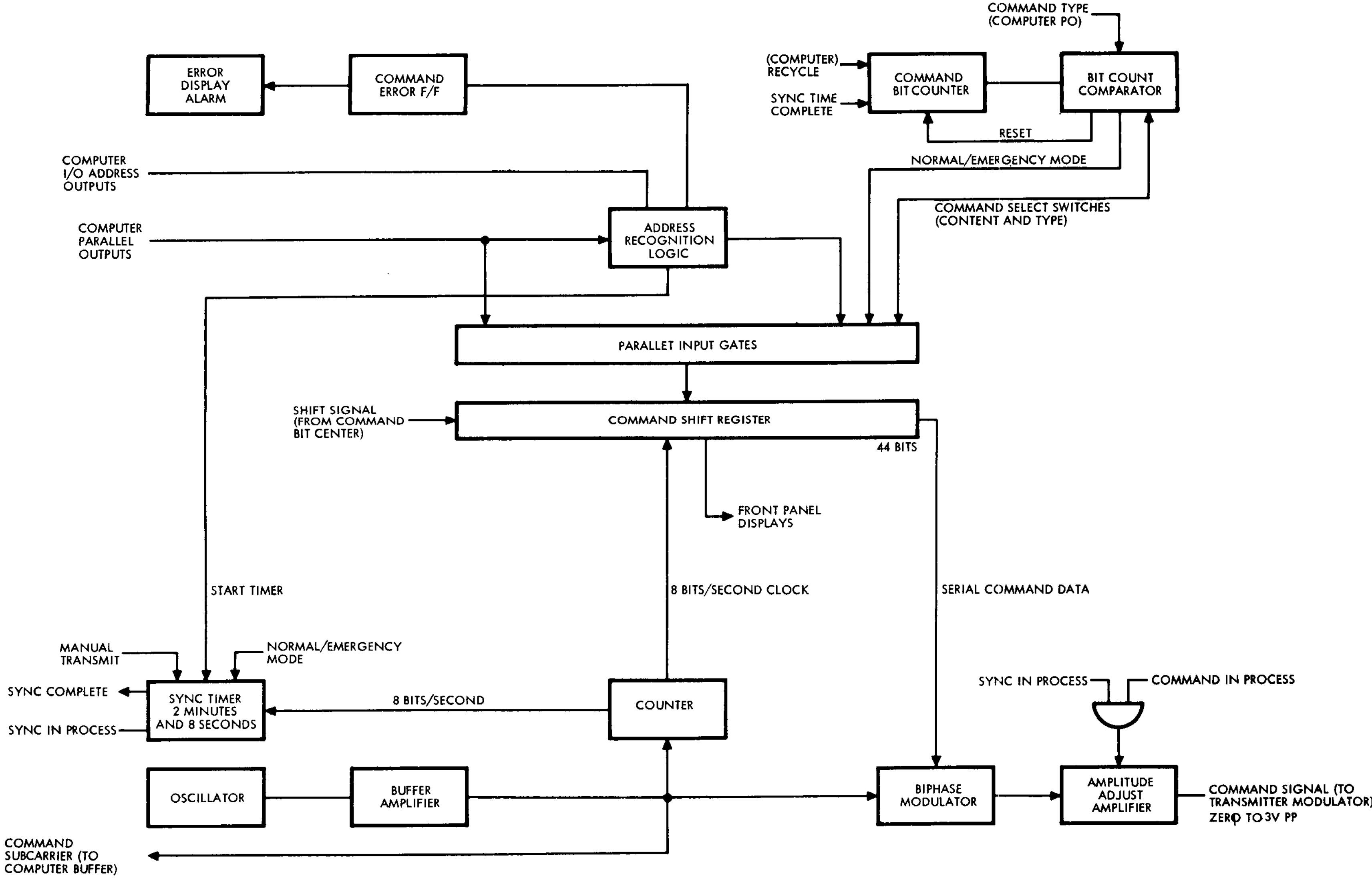


Figure 6-10
COMMAND ENCODER generates planetary vehicle commands which modulate the DSS command transmitter frequency.

module, card mounted. The oscillator output is amplified to drive the modulator circuit. With the subcarrier and the serial command data as inputs, the biphase modulator changes the phase of the subcarrier signal by 180 degrees each time a "one" appears in the serial command data. The amplitude adjust amplifier is an isolation amplifier that amplifies the modulator output, provides isolation between the command encoder and the transmitter/modulator, and allows for amplitude adjustment of the output from zero to approximately 3 volts peak-to-peak. The "or" circuit input to this amplifier allows for the modulator output to be fed through the amplifier only when a command is in process or the station/spacecraft command link is being synchronized (approximately 2 minutes).

The function of the command storage and display section is to provide for the selection, storage, and display of a single command from either the TCP computer (normal mode) or from front panel selection switches (emergency mode). The command register is a flip-flop register that is parallel loaded via input gates from either the computer or the selection switches. A front panel switch signal (normal/emergency) determines the source of command data to be loaded into the register. Address recognition logic is included to logically recognize when the parallel input gates should be enabled to allow the computer to load command data into the register. After the command is loaded into the register, the register is shifted by an 8 bits/sec clock and a shift enable signal is provided by the logic control section. For all three types of planetary vehicle commands, the same set of nixie tubes, will be utilized for the display of elements A, B, P_1, C, and P_2. Command message elements D, E, and F will be displayed separately. The logic control section generates several timing signals and provides the supervisory control for the following functions performed internal to the unit:

- Derives the 8 bits/sec clock
- Derives the 2 minute synchronization signal
- Generates computer interrupts for command transmission verification checks.

The 8 bits/sec clock, which establishes the command data rate, is derived by counting down the subcarrier signal. The 2-minute command link synchronization signal is derived by counting down the 8 bits/sec clock, after the counter is started by either the computer or manual transmit signal. This signal is used to enable the unmodulated subcarrier output for a 2-minute period to allow the command subsystem to synchronize to the station, prior to command transmission. The command shift signal is derived from the command bit counter and its associated count comparator. This circuitry specifies the length of time the command register must shift at the 8 bits/sec rate to complete the type of command lengths. The computer interrupt is generated at the middle of each bit time during a command transmission to signal the computer when data is available for verification of proper transmission. The interrupt signal for the last command data bit can also signal the computer to immediately (within 62 ms) load the next command, in order to avoid having to repeat the 2-minute synchronization procedure. In addition to the above, this section also contains the control logic necessary for testing of the station command loop and self test of the command encoder unit. For the station test, bit error will be automatically injected to verify the bit-by-bit comparison check performed by the TCP computer. Self test of unit consists of verification of the command message selection and display functions performed at a special bit rate of 1 bits/sec.

6.6.3 Physical Characteristics

The command encoder unit will be a slide-mounted drawer assembly containing plug-in integrated circuit cards. The slide-mounted assembly is a standard TRW Systems design for ground support equipment. The plug-in digital circuit modules have been developed by TRW Systems for application on all deliverable ground support equipment. This circuit line is designed to meet the requirements of MIL-STD-810 and is capable of operation within an ambient environment from 0 to +55°C. The integrated circuit elements are of the diode-transistor-logic type.

6.6.4 Performance Requirements

a) Design Parameters

Command Output Signal

Subcarrier Frequency:	cps ±0.02 percent
Distortion:	less than 2 percent
Amplitude:	adjustable zero to 3 volts peak-to-peak isolated output into 50 ohms
Amplitude Stability:	1 percent over 2 hour period
Bit Rate:	8 bits/sec ±0.003 percent

b) Signal Inputs and Outputs

Inputs

Computer parallel outputs
Computer address outputs
} from computer buffer (MDE)

+12 VDC
+6 VDC
+250 VDC
} from power supply (MDE)

Outputs

Spacecraft command signal
Unmodulated subcarrier
} to transmitter modulator (MIE)

Computer interrupt
Subcarrier
Simulated monitor receiver output
} to computer buffer (MDE)

c) Displays

Command content

Sync in process

Command error

Transmit

Power on

d) Controls

Command selection switches
Command type selection
Transmit switch
Power on

6.7 COMPUTER BUFFER

6.7.1 Functional Description

The primary function of the computer buffer is to provide a flexible capability of collecting serial telemetry data and inserting this data into the station computer in parallel groups convenient for computer handling. In addition, this unit also serves as a communication path for the transfer of information between the computer and the communications processor, associated station status displays, the capsule contractor equipment and the command encoder. This unit is also capable of demodulating the output of the monitor receiver (modulated command subcarrier) during all command transmission, or a simulated test signal from the command encoder during station test.

6.7.2 Design Implementation

The design of the computer buffer as shown in Figure 6-11 is organized into the following three sections: data collection, data insertion and data distribution. The data collection section consists of telemetry data selection logic and a telemetry data shift register. The telemetry data selection logic allows for selecting the source of telemetry from either the demodulation/synchronizer, the biorthogonal decoder, or the data format generator. The inputs consist of either uncoded or decoded data, a data clock, and a synchronization status discrete signal. The selection of the source of data can be either manual or under computer control.

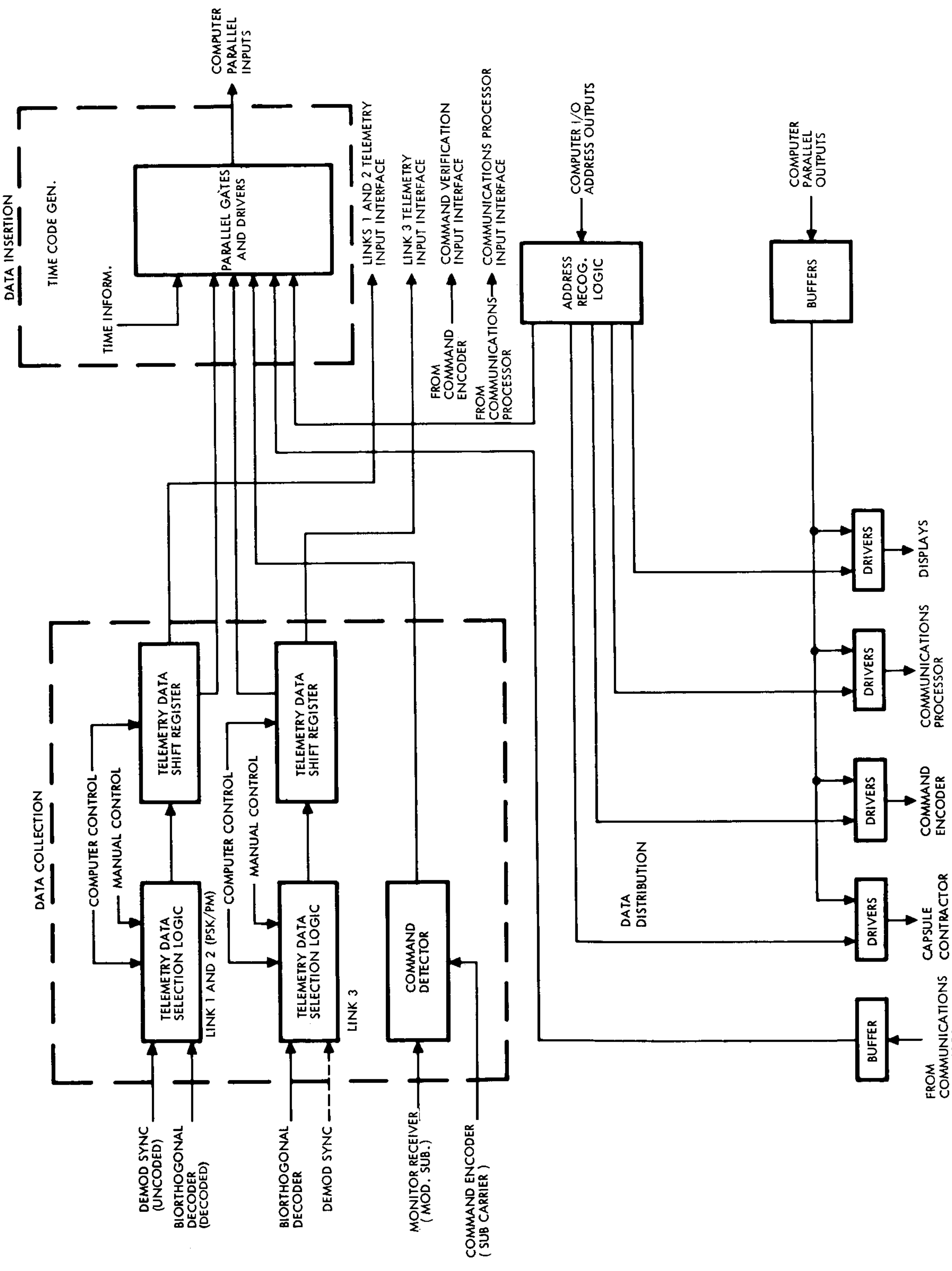


Figure 6-11
COMPUTER BUFFER serves as a communication media for transfer of information between the computer and MDE/MIE.

The telemetry data shift register collects the serial telemetry data and applies the data to the data insertion section for transfer into the computer. The shift register is partially under the control of the computer, such that the computer can specify the number of parallel data bits or words to be inserted into the computer on each transfer. The shift register generates a data ready signal to signal the computer that the specified amount of data is available for insertion.

Two sets of telemetry data selection logic and data shift registers are provided to handle the two simultaneous telemetry links from one spacecraft. In addition, a command detector is included to detect the monitor receiver output and convert this signal into binary data for transfer to the computer for command transmission verification.

The data insertion section serves the function of inserting the data from the following sources into the computer:

- Telemetry links 1, 2 or 3
- Command detector output
- Communications processor data
- Time information.

The telemetry data, command detector data, and the communication processor data are inserted under the control of the computer, after having acknowledged the input ready signal (interrupts) generated by the data source. The address recognition logic allows the buffer to enable the proper logic gates to transfer the data requested by the computer. The time data can be sampled by the computer as required.

The data distribution section serves the function of routing computer output data to the following areas:

- Station status displays
- Communications processor
- Command encoder
- Capsule contractor MDE.

It is assumed that all computer output data can be distributed to these users in parallel form. The implementation requires that the computer first address the unit to which data is to be transferred and subsequently transfer the data via its parallel outputs. The address recognition logic enables the output drivers to the unit to receive the computer data and the data is transferred to the unit via the output drivers.

6.7.3 Physical Characteristics

The computer buffer unit will be a slide-mounted drawer assembly containing plug-in integrated circuit cards similar to those described for the command encoder unit.

6.7.4 Performance Requirements

a) Signal Inputs and Outputs

1) Inputs

Demodulator/synchronizer Link 1 and 2
Demodulator/synchronizer Link 3
Biorthogonal decoder Link 1 and 2
Biorthogonal decoder Link 3
Monitor Receiver
Command Encoder
Communications Processor
Station Computer
Time Code Generator
Data Format Generator
Power (DC)

2) Outputs

Station Computer
Station Displays
Communications Processor
Command Encoder
Capsule Contractor MDE
Digital Instrumentation System via Communications Processor

b) Displays

Power On

Audible Alarm (connect/disconnect)

c) Controls

Telemetry Source Selection

Audible Alarm

Power On/Off

6.8 PLANETARY VEHICLE SUBSYSTEM DISPLAY

The planetary vehicle subsystem display (Figure 3-3) is a computer driven display consisting of three standard 19-inch racks. The computer monitors the downlink telemetry data and then makes decisions as to the general status of the spacecraft. The status information is routed to the display. The planetary vehicle subsystem status display provides information about the operating modes of the various vehicle subsystems, indicates the occurrence of major events, indicates malfunctions, and displays certain important engineering measurements.

6.8.1 Communication Subsystem

The operating mode of the communication subsystem is depicted in flow diagram on the front panel of this unit. The flow diagram indicates which transmitters and receivers are operating and also indicates the selected antennas. The computer monitors the following telemetry signals:

- Baseband output
- Exciter No. 1 power output
- Exciter No. 2 power output
- 1-watt transmitter output
- Transmit selector mode
- Receivers Nos. 1, 2, 3 and 4 in-lock
- Receiver selector mode
- Low gain antenna selector mode
- Circulator assembly mode

6.8.2 Propulsion module

The status display for the propulsion module indicates when the engine is operating and whether it is operating in the high thrust mode or the low thrust mode. The computer monitors the engine chamber pressure, position of the four quad solenoid valves, and other parameters of interest.

6.8.3 Guidance and Control Subsystem

General status of the guidance and control subsystem is displayed by indicating power on/off, the mode of operation, acquisition of the sun, acquisition of Canopus, and gas bottle pressures and temperatures. The computer monitors the following telemetry signals:

- Mode control monitor
- Canopus sensor sun present
- Canopus sensor star present
- System A gas bottle pressure
- System B gas bottle pressure
- System A gas bottle temperature
- System B gas bottle temperature

6.8.4 Telemetry Subsystem

The operating modes of the telemetry subsystem will be displayed including data transmission bit rates and symbol rates and type of information being transmitted, e.g., video, engineering or science data.

6.8.5 Data Handling Subsystem

The status displays for the data handling subsystems indicate power on/off and recorder operation. The telemetry data from the recorders are monitored to determine the mode of operation (read/write) and whether or not the tape is in motion.

6.8.6 Electrical Power and Distribution Subsystem

The status display for the electrical power and distribution subsystem indicates the general status of the solar arrays, the batteries, the 400 Hz inverters, the power control unit, the distribution power control unit, and the pyrotechnics control unit. The following telemetry signals are monitored.

- Power control unit voltages
- Battery voltages
- 400 Hz inverter voltages
- Solar arrays currents
- DCU supply voltage
- Pyro control supply voltage
- Subsystem temperatures of interest

6.8.7 Command and Sequencing Subsystem

The general status indicates power on/off for the computer/ sequencer unit and the command unit and bit sync lock for the command unit. Command events and status information of both the computer/ sequencer units and the command units from telemetry are displayed.

6.8.8 Pyrotechnics and Thermal Control Subsystem

The safe-arm condition of the pyrotechnics is indicated along with the major thermal control parameters such as louver positions. The following telemetry signals from the pyro control unit will be monitored:

- Safe-Arm 1
- Safe-Arm 2
- Safe-Arm 3
- Safe-Arm 4
- Safe-Arm 5

6.8.9 Science Subsystem

In the science subsystem indications will be given as to which experiments and which cameras are operating. This information will be derived by monitoring telemetry data from the science subsystem and the command and sequencer subsystem.

6.9 DC POWER SUPPLY

The DC power supply unit will supply various output voltages as required by the associated rack of equipment. The power supplies primary input voltage is 115 ±10 VRMS at a frequency of 60 ±5 Hz. Each of the various low voltage output voltages contains the following features:

- Voltage Control: Adjustable by a front panel control over a narrow voltage range.
- Current Limiting: Adjustable to limit the output current such that a short circuit will not damage the power supply.
- Overvoltage Protection: Adjustable overvoltage "crow-bar" circuit which shorts out the power supply output and trips the power supply circuit breaker when the power supply output voltage exceeds the set level. The overvoltage is reduced to less than 4 VDC within 50 microseconds after the overvoltage occurs.

The front panel contains a power control push-button indicator type switch to turn off/on all the rack DC power supplies at one time. A DC meter with appropriate scales and in conjunction with a rotary switch permits monitoring of each DC output voltage. Meter accuracy is ±5 percent or better.

All power supply assemblies used with units constructed of standard logic cards will be identical for both MDE and EOSE equipment. This will provide cost effective design and manufacture as well as reduce the number of spare power supply assemblies required.

6.10 INTEGRATION, HARDWARE, CABLES, RACKS, AC POWER CONTROL

6.10.1 Cables

Functional Description. The MDE electrical harnesses and cabling are assemblies constructed to distribute all power and signals for interconnecting all MIE and MDE subsystems. The harness and cabling assemblies consist primarily of multiple insulated wires terminated to various connectors and secured by means of spot ties. All shielding, shielding terminations, and grounding will be accomplished with particular emphasis on eliminating EMC problems.

Design Considerations

a) Materials

Interconnecting harness materials will conform to the following minimum requirements:

1) Suitable materials will be used to minimize permanent induced and transient magnetic fields

2) Materials will not be radioactive

3) Materials will be radiation resistant

b) Shielding Termination

Shields will be terminated by one of the three following methods and are covered within TRW specification PR7-3B.

1) Use of heat-shrinkable solder-sleeves as per referenced TRW specification and NASA specification SR-QUAL-65-25

2) Use of Thomas and Betts inner and outer shield grounding sheaths on multiple shield terminations.

3) Use of "Halo-Ring" shield grounding adaptor on "D" series connectors.

Application of one of the methods indicated above will be determined during design.

c) Connectors

In order to adhere to the requirements of the specifications and to take advantage of off-the-shelf components, the following connectors are recommended for use on the electrical harness:

- 460/466 (Deutsch) - Connectors, circular, environmental resisting, general purpose with removable crimp contacts. These are intermateable with all NAS-1599, MIL-C-26482 and MIL-C-26500 bayonet coupling connectors.

- Royal Mark III - D-series rack and panel type connectors. Crimp, removable contacts

The performance of the above suggested connectors in past and present programs has resulted in the connector being selected as the TRW standar miniature circular connector.

The use of removable crimp-contact-type connectors will result in a harness that is significantly less difficult to assemble and inspect and will aid in producing a higher-reliability, lower weight harness. The dimensions detailed in Figure 6, page 7, of NASA document SR-QUAL-65-25 will be used as a design guide for shielded wire terminations.

d) EMI Consideration

The achievement of an electromagnetically compatible subsystem requires a uniform and consistent systems approach throughout all phases of design, development, and testing. The integration of a number of components requires systematic control of the electromagnetic compatibility considerations to ensure that the components operate individually and collectively without malfunction or degradation in performance due to electromagnetic interference.

The initial phase of the program will consist of subsystem component analysis and compatibility prediction to establish EMI control techniques. As a goal, the analysis and prediction will include the following parameters:

- Power and signal interface characteristics
- Signal and power circuit grounding constraints
- Circuit susceptibility thresholds
- EMC status of GFE
- Thermal - electrical bonding constraints
- Sensitive frequency bands
- Operational frequencies and levels
- Special test requirements.

The results of these analyses will form the basis for developing the design and test requirements for the electrical harnesses and cables of the DSS.

e) Continuity

Each conductor will be tested for continuity and correct connections between its terminations.

f) Insulation Resistance

Each conductor will show a minimum resistance of 100 megohms between the conductor and every other conductor and shield. The applied megger potential shall be 500 v ±25 v.

g) Mechanical

Visual inspection of the harnesses and cables will be performed to detect mechanical defects prior to installation or test.

6.10.2 Racks

The MDE electronic equipment enclosures will be standard JPL specified racks suitably modified. The modifications will include any special provisions required for EMI control, cooling air circulation and equipment mounting.

6.10.3 AC Power Control

The AC power supply drawers receive the AC primary power. The input power passes through a magnetic trip circuit breaker, a set of relay contacts and line filters into plug-in strips.

A power control push button indicator type switch controls the set of relay contacts such that all of the primary power can be turned ON or OFF with one switch. A running time meter and blower fuse indicator lamp complete the drawer.

7. MDE SOFTWARE INTRODUCTION

The mission-dependent computer programs will be integrated into those existing SFOF, HOSC, DSIF, and TRW data processing systems to enable mission operation teams to monitor the performance of the spacecraft system and experiments; to process and correlate information required for engineering evaluations and operational decisions; to formulate alternative courses of action and evaluate the implications of each upon mission success; and to select and implement the best course of action from those possible.

The development of the family of mission-dependent computer programs begins during the early mission analysis and design phase and will be formalized during the implementation phase. The primary assumption inherent within the computer programs described is that they will be used with computer systems to be built and operating in the mid-1970's, and hence may require processing capabilities beyond the current state of the art.

One such program is the Spacecraft Response Simulation Program. It represents a significant advancement in capability for preflight analysis and training and in-flight evaluation of a spacecraft's performance over any program in existence. This program will simulate the response of the spacecraft to transmitted commands or on-board subsystem failures, and thereby assist in the formulation of contingency plans. It can compare the effects of alternative operation sequences and verify subsystem functions and their interactions. It will also be utilized to perform preflight mission planning and feasibility studies, to train operational personnel in system monitoring and performance evaluation, and to perform operational readiness tests and system checks to ensure inter-systems compatibility.

All of the programs are described in the form of a software specification. Where a program used in a simulation function is nearly identical with the program used in the monitoring function, one program is described with their differences delineated. Similarly, only the differences in program utilization between SFOF and HOSC are discussed, since the functions performed are so much alike.

7.1 SOFTWARE FUNCTIONS

The SFOF software system is comprised of programs that can be subdivided into four categories:

- SFOF on-line real-time system
- SFOF simulation and support system
- SFOF simulation and training system
- Spacecraft Response Simulation Program.

7.1.1 SFOF On-Line Real-Time System

a) Related Programs

- SFOF Resident Executive
- Telemetry Processing Program
- Electric Power Subsystem Monitoring Program
- Thermal Control Subsystem Monitoring Program
- Propulsion Module Monitoring Program
- Guidance and Control Subsystem Monitoring Program
- S-Band Radio Subsystem Monitoring Program
- Spacecraft Status Display Program
- Science Instrumentation Performance Monitoring Program
- Science and Video Command Generation Program
- Command Composition Program
- Command Verification Program

b) Functional Description. The SFOF real-time system of computer programs supports the spacecraft performance analysis teams in monitoring spacecraft performance by means of automatic alarm procedures and displays. They will:

- Monitor spacecraft system performance
- Process, correlate, and handle information required to perform engineering evaluations
- Formulate alternative courses of action and evaluate the implications of each upon mission success.

The SFOF real-time system is illustrated in Figure 7-1.

c) Interface Definitions. Program interfaces for the SFOF computer programs are provided in each detailed functional description.

7.1.2 SFOF Simulation and Support System

a) Related Programs

- SFOF Resident Executive Program
- Electric Power Subsystem Simulation Program
- Thermal Control Subsystem Simulation Program
- Propulsion Module Simulation Program
- Guidance and Control Subsystem Simulation Program
- S-Band Radio Subsystem Simulation Program
- Telemetry Processing Station Program
- Video Enhancement Program
- Video Processing Program
- Maneuver Analysis and Command Program

b) Functional Description. The SFOF simulation and support system is comprised of a set of computer programs which are designed to support the SFOF on-line, real-time system by providing the simulation interfaces required by the subsystem monitor programs; to support mission planning and command generation activities by providing full or partial mission simulations of respective subsystems; and to support the simulation and training system by providing simulated data for use in exercising the entire system and in the training of personnel (see Figure 7-2).

c) Interface Definitions. Interface specifications for the SFOF simulation and support system computer programs are provided in their respective functional specifications.

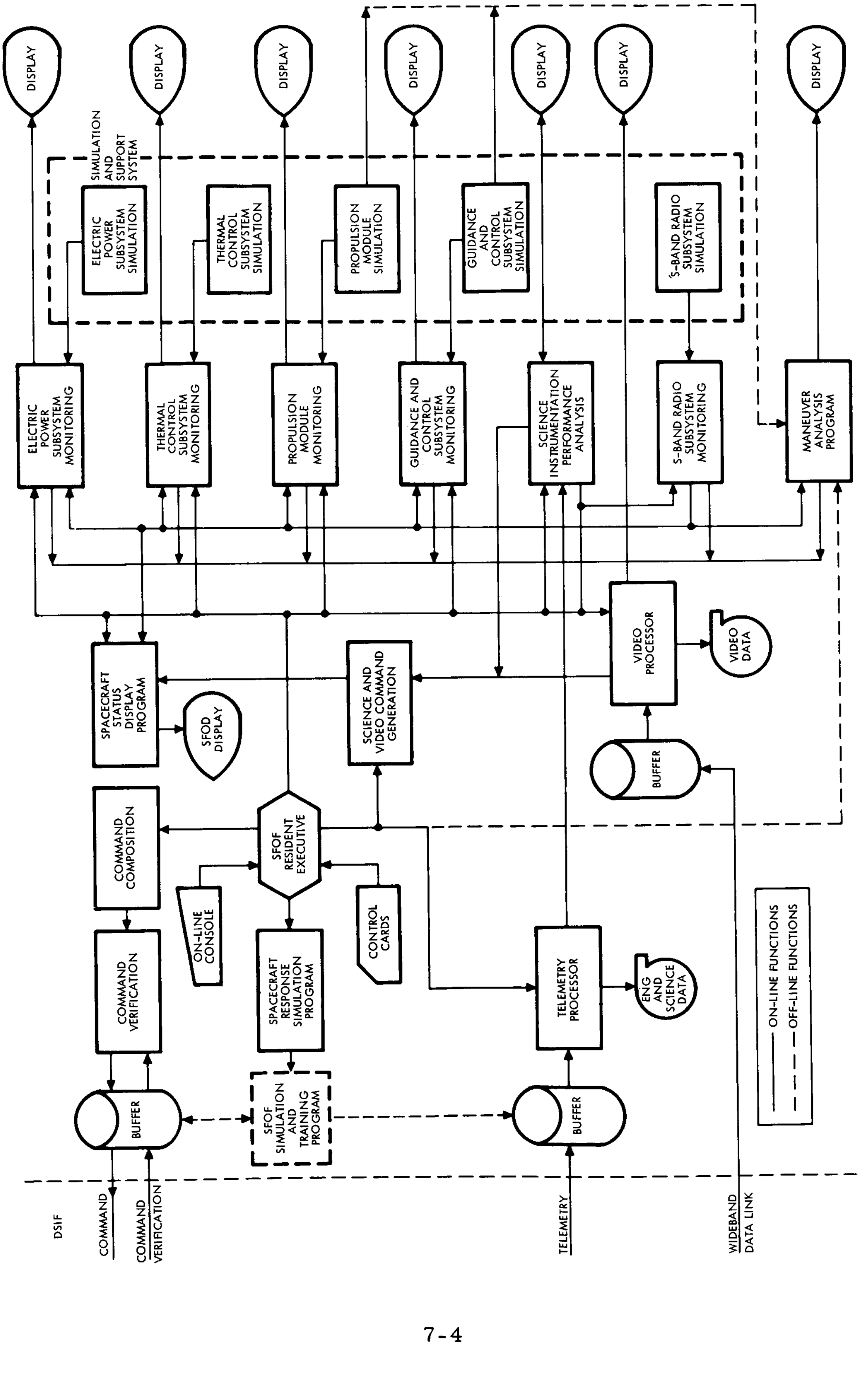


Figure 7-1
SFOF MONITORING AND SIMULATION programs are interfaced to provide optimum training, checkout and inflight capabilities.

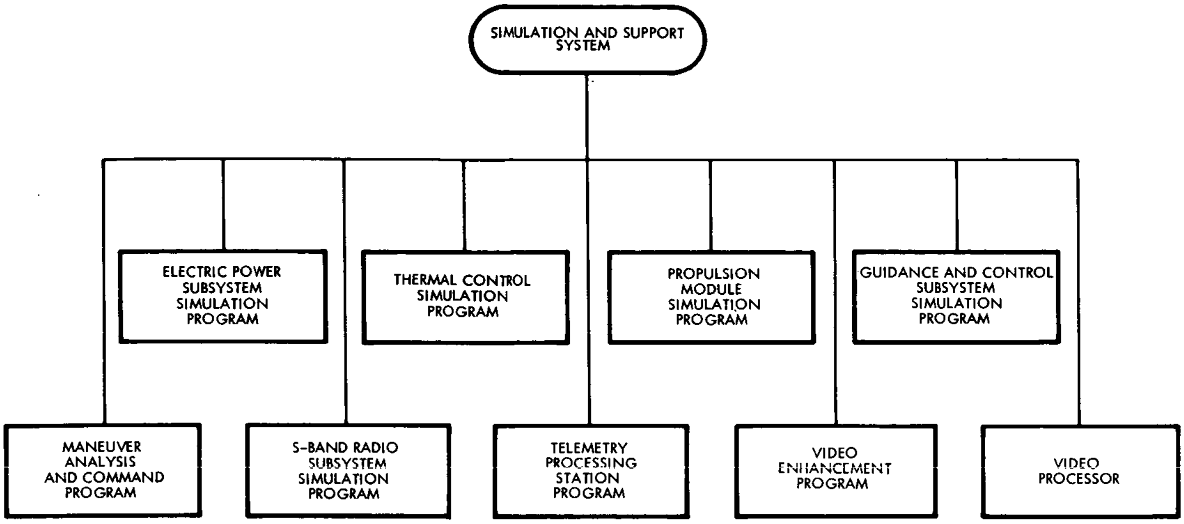


Figure 7-2
SFOF SIMULATION AND SUPPORT SYSTEM off-line computer programs handle interfaces with various spacecraft system and subsystem simulation programs.

The programs comprising the system are described in the following specifications:

Telemetry Processing and Processing Station Programs

a) Related Programs

- SFOF Resident Executive Program
- Electric Power Subsystem Monitoring Program
- Thermal Control Subsystem Monitoring Program
- Propulsion Module Monitoring Program
- Guidance and Control Subsystem Monitoring Program
- SFOF Simulation and Training Program
- S-Band Radio Subsystem Monitoring Program
- Spacecraft Status Display Program

b) Functional Description. The Telemetry Processing Program will receive the telemetry data from the deep space stations via high speed data link and disk storage. It will:

- Assemble the data stream into complete frames
- Check for proper frame synchronization, continuous timing, and recognition of data dropout periods

- Convert vehicle clock to ground time
- Separate engineering from scientific data
- Produce a binary tape containing engineering data
- Convert engineering data to appropriate units required as input to the monitoring and simulation programs
- Detect discrete status changes.

The Telemetry Processing Station Program will accept recorded or simulated telemetry data and perform the above functions. In addition, it will:

- Produce a binary tape containing scientific data
- Convert scientific data to appropriate units required for distribution to the experimenters.

During nominal and peak activity periods, the SFOF telemetry processing station off-line program will operate with the on-line decoded serial data received by the SFOF on-line, real-time system through the data line interface with the DSIF. Non-nominal operations will employ delayed digital tapes or transmissions from the DSIF. In the event that the decoded serial data are not available, the system will extract the necessary data from the DSIF analog recordings through the SFOF demodulation and/or decoding equipment.

c) <u>Interface Definitions</u>

- <u>Telemetry Processing Program Inputs</u>. Data from on-line telemetry buffer; data from a disk containing telemetry data received from the deep space stations or from tape generated by the SFOF Simulation and Training Program.
- <u>Telemetry Processing Station Program Input.</u> Raw telemetry data or simulated telemetry data provided by the SFOF Simulation and Training Program.

  Run parameters and control parameters will be entered on punched cards.

  Calibration data will be input from magnetic tape or disk.

External data will be input from magnetic tape or disk.

Telemetry data will be input from magnetic tape or disk.

- Telemetry Processing Program Outputs. Binary tape containing all engineering data for record keeping; binary tape containing all scientific data for record keeping and subsequent distribution; disk data for use by other programs; on-line output regarding the quality of the telemetry data.

- Telemetry Processing Station Program Output

  Calibrated telemetry data and external merged data will be output via magnetic tapes.

  Discrete events, statistical data, data quality information, and program diagnostics will be given as printer output.

  Binary tape containing all engineering data for record keeping

  Binary tape containing all scientific data for distribution and record keeping

  Disk and tape data for use by other programs

  Output regarding the quality of the telemetry data.

Electric Power Subsystem Monitoring and Simulation Programs

a) Related Programs

- SFOF Resident Executive Program
- Telemetry Processing Program
- Spacecraft Status Display Program

b) Functional Description. The monitoring program monitors the parameters that characterize battery, solar array, and power control unit performance.

The power system performance analysis team is informed of the status of the system via display, on-line printout, and/or on-line plots. The program compares real-time power system telemetry data with simulated data from the Electric Power Subsystem Simulation Program. If the characterizing parameters exceed specified tolerance limits, automatic alarm messages are printed. Based on the output of this program, the power system performance group will recommend to the SPAC director the appropriate action and/or command requests.

The simulation program determines the power utilization of all spacecraft subsystems and calculates the parameters characterizing battery and solar array status and determines the most effective mode of converting, regulating, and distributing power to all subsystems.

c) Interface Definitions

- Monitoring Program Inputs. Telemetry data in engineering units from the Telemetry Processing Program; predicted power system performance parameter values from the Electric Power Subsystem Simulation Program.

- Simulation Program Inputs. Program input will be in the form of cards, tape, and/or disk. Run data and output control data will be input to the program on cards. All initialized parameters and table alterations will be changed by cards. Modifications to the mission profile or time line would also be accomplished through card input. Additional timeline data and mission profile data will be input via tape or disk.

- Monitoring Program Outputs. Power system performance parameter printouts, plots, and/or displays; alarm messages and status change printouts displays; parameters characterizing the operational status of the following power subsystems will be displayed: power control unit, battery, 400-Hz inverter, and solar array.

- Simulation Program Outputs

Tabulated output

Initial conditions	Total power loss/gain
Parameter tables	Regulated power
Start time (t_o) and stop time (t_s) of interval	Unregulated power
System constraint violations	Total power used

Out-of-tolerance data	Battery terminal voltage
Power failures	Battery current
Battery status	Nodal power conditions
Solar array status	All power loads on between times t_n and t_m
Battery power loss/gain	All calculated data point values between t_n and t_m
Solar panel power loss/gain	

Tape/disk output

All calculated and input quantities required for support of other programs and systems will be output on tape or disk in an established compatible format.

Thermal Control Subsystem Monitoring and Simulation Programs

a) Related Programs

- SFOF Resident Executive Program
- Telemetry Processing Program
- Spacecraft Status Display Program

b) Functional Description. The monitoring program monitors spacecraft temperature data and determines thermal gradients and heat flow parameters. The analysis teams are informed of the status of the system via display, on-line printouts, and/or on-line plots.

A comparison is also made between the thermal control performance and established design criteria as well as simulated temperatures. The predicted temperatures are obtained from the Thermal Control Subsystem Simulation Program. If temperatures exceed specified limits, automatic alarm messages are printed. Based on the output of this program, the SPAC director will take the appropriate action and/or make the appropriate command requests.

The simulation program determines and predicts temperatures of all spacecraft components, calculates thermal gradients and heat flow parameters, compares the thermal control performance against established design criteria, and simulates thermal profile for analysis of eclipse conditions. In addition, the program will have the capability of computing steady-state temperatures.

c) Interface Definitions

- Monitoring Program Inputs. Telemetry data in engineering units from the Telemetry Processing Program; predicted temperatures from the Thermal Control Simulation Program.

- Simulation Program Input. Input to the program will be in the form of cards, tape, and/or disk. Run data and output control parameters will be input to the program on cards. All initialized parameters and table alterations will be changed via cards. Orientation and nodal power time-line data and other mission profile data will be input via tape or disk.

- Monitoring Program Outputs. Temperature printouts, plots, and/or displays of temperature and performance parameters for at least 40 spacecraft locations; alarm messages and status change printouts.

- Simulation Program Output

  Typical Tabulated Output:

  Initial conditions

  Parameter tables

  Temperature versus time data

  Out-of-tolerance data

  Indication of thermal constraint violations

  Node temperature versus time

  Tape/Disk Output:

  All calculated and input quantities required for support of other programs and systems will be output on tape or disk in an established compatible format.

Propulsion Module Monitoring and Simulation Programs

a) Related Programs

- SFOF Resident Executive Program

- Telemetry Processing Program

- Spacecraft Status Display Program

b) Functional Description. The monitoring program monitors propulsion module performance parameters and computes a time history of fuel expenditures. The analysis teams are informed of the status of the system via display, on-line printout, and/or on-line plots. The program compares actual performance parameters and predicted performance parameters determined by the Propulsion Module Simulation Program. If these parameters exceed specified limits, automatic alarm messages are output. Based on the output of this program, the propulsion module performance group will recommend to the SPAC director the appropriate action and/or command requests.

The simulation program generates predictions of the performance of the propulsion module and budgets fuel expenditure of the propulsion and reaction control systems. In addition, it integrates fuel expenditures throughout thrusting periods and accounts for losses by venting and leakage.

c) Interface Definitions

- Monitoring Program Inputs. Telemetry data in engineering units from the Telemetry Processing Program; predicted performance parameters from the Propulsion Analysis Simulation Program.

- Simulation Program Inputs. Input to the program will be basically via cards; however, depending on the operational environment, it could prove desirable to input thrust command information from tape or disk.

- Monitoring Program Outputs. Propulsion module parameters printouts, plots, and/or displays; alarm messages and status change printouts. The characteristic output parameters include pressures, temperatures, and bi-level event status messages for the pressurization, feed, and engine systems.

- Simulation Program Output. The basic output from this program will consist of a tabulated accounting of all discrete expenditures and a cumulative account of expenditure, total remainder, and reserve allocations. In addition, overexpenditures and reserve critical points are identified and propulsion module performance parameters are provided.

Guidance and Control Subsystem Monitoring and Simulation Program

a) Related Programs

- SFOF Resident Executive Program
- Telemetry Processing Program
- Spacecraft Status Display Program

b) Functional Description. The monitoring program computes the spacecraft guidance and control subsystem performance parameters. The analysis teams are informed of the guidance and control subsystem status via display, on-line printouts, and/or on-line plots. The program compares predicted with actual performance parameters and enacts the automatic alarm if tolerance limits are exceeded. Based on the output of this program, the guidance and control subsystem performance group will recommend to the SPAC director the appropriate action and/or command requests.

The simulation program computes predictions of performance parameters of the spacecraft guidance and control subsystem including parameters that characterize the operational status of gyros, sensors, accelerometers, and thrust vector control system. For each of the mission phases, the program will compute pointing angles of the earth tracking stations; Mars, the sun and Canopus in vehicle; earth- and Mars-centered coordinate systems.

c) Interface Definitions

- Monitoring Program Inputs. Telemetry data in engineering units from the Telemetry Processing Program; predicted performance parameters from the Guidance and Control Simulation Program.

- Simulation Program Inputs. The program input will be in the form of cards and magnetic tape or disk files. Initialized parameters, run and execution data, output control data, and table alternations will be input by means of punched cards. Data generated by other programs, such as trajectory data, will nominally be input by means of tapes or disk. However, where desirable, the program will also accept this type of data from cards.

- Monitoring Program Outputs. Printouts, plots, and/or displays of guidance and control subsystem performance parameters; alarm messages and status changes; the parameters that characterize the operational status of gyros, sensors, accelerometers, and thrust vector control systems.

- Simulation Program Output

  Typical Tabulated Output:

  Initial conditions

  Parameter tables

  Coordinate system designation

  Angular orientation time histories

  Interval time

  Constraint violations

  Gyro position and rates (pitch, roll, yaw)

  Sun, Canopus, near-earth, and limb crossing terminator sensor parameters

  Thrust vector control actuator parameters

  Gas bottle pressure parameters

  Tape/Disk Output:

  All calculated and input quantities required for support of other programs and systems will be output on tape or disk in an established compatible format.

S-Band Radio Subsystem Monitoring and Simulation Programs

a) Related Programs

- SFOF Resident Executive Program
- Telemetry Processing Program
- Spacecraft Status Display Program

b) Functional Description. The monitoring program will monitor all transmitter, modulator exciter, power amplifier, S-band receiver, receiver selector, transmitter selector, RF switch, and capsule relay link parameters. The program compares real-time data with prediction data furnished by the S-Band Radio Subsystem Simulation Program. If the parameters characterizing this system exceed specified limits, automatic alarm messages are provided. Based on the output of this program, the SPAC director will take the appropriate action and/or make command requests.

The simulation program computes nominal DSIF-received signal-to-noise ratios as a function of trajectory, ground system, and space vehicle component parameters; determines anticipated signal strength at the DSIF and spacecraft based upon range, space orientation, antenna pointing, effective radiated power and other parameters; and generates communication forecasts and bit error rate forecasts for transmission to DSIF stations. Predictions of the S-band radio subsystem performance will be compared with actual flight-generated data.

c) Interface Description

- Monitoring Program Inputs. Telemetry data in engineering units from the Telemetry Processing Program; prediction data provided by the S-band Radio Subsystem Simulation Program.

- Simulation Program Input. Input to the program will be in the form of cards, keyboard, and tape or disk. Run data, output control parameters, and mode data will be input or modified by punched cards and/or keyboard. Trajectory data, AGC data, and antenna pattern data will be input from tape and/or disk.

- Monitoring Program Outputs. Printouts, plots, and/or displays of S-band radio subsystem performance parameters; alarm messages and status change printouts.

- Simulation Program Output

  Tabulated Output:

  Listing of input control statements and telecommunication parameters. An ordered listing for the spacecraft and each DSIF station indicating the overall mode of the receiver carrier power, the computed performance margins, related tolerances, and the parameter value used for the computation at a fixed range and antenna look angle.

  A listing for the spacecraft and each DSIF station of the modes of received carrier level of the performance margins and of the tolerance as a function of time.

Plotted Output:

The program shall plot the predicted and measured received carrier levels for both earth-to-spacecraft (E-S) and spacecraft-to-earth (S-E) S-band radio channels, as well as the AGC calibration curve for each DSIF station.

A composite plot of indicated optimum performance with time for each station and mode will be generated as a guide for data selection.

Spacecraft Status Display Program

a) Related Programs

- SFOF Resident Executive Program
- Telemetry Processing Program
- Electric Power Subsystem Monitoring Program
- Thermal Control Subsystem Monitoring Program
- Propulsion Module Monitoring Program
- Guidance and Control Monitoring Program
- S-Band Radio Subsystem Monitoring Program

b) Functional Description. This program will drive the displays which present the status of all spacecraft subsystems. Aside from the spacecraft subsystems for which on-line monitoring is performed, status information for the computer and sequencer, data recorders, telemetry, command, pyrotechnic, and video subsystems will be provided. In addition, spacecraft trajectory-related parameters will be displayed. The output of this program aids the user area personnel and the mission director in their selection of appropriate action.

c) Interface Definitions

- Program Inputs. Telemetry data in engineering units from the Telemetry Processing Program; status information from the subsystem monitoring programs; status information relating to the spacecraft trajectory from Maneuver Analysis and Command Program.

- Program Outputs. Display data characterizing spacecraft system performance; appropriate alarm messages; trajectory-related information; data defining the mode in which the subsystems are operating.

Science Instrumentation Performance Monitoring Program

a) Related Programs

- SFOF Resident Executive Program
- Science and Video Command Generation Program
- Spacecraft Status Display Program
- Telemetry Processing Program

b) Functional Description. The functions of this program are to monitor and analyze the performance of the science instrumentation. The program:

- Selects from the processed telemetry data the significant parameters pertinent to each of the science experiments
- Monitors these parameters and compares them with predicted values
- Displays the time history of these significant parameters
- Generated alarm messages whenever the monitored parameters exceed preassigned tolerance limits.

c) Interface Definitions

- Program Input. Science instrumentation performance data from the telemetry processing program
- Program Output. Science instrumentation performance parameters relating to the following experiments:

Video	Neutron albedo
High-resolution infrared spectrometer	Magnetometer
Broadband infrared spectrometer	Micrometeoroid impact detector
Infrared radiometer	Flash
Neutron velocity spectrometer	
Atmospheric mass spectrometer	Interplanetary particle and solar x ray

Polarimeter	LEPEDEA
Solar occultation	X ray
Gamma-ray spectrometer	Celestial x ray

Science and Video Command Generation Program

a) Related Programs

- SFOF Resident Executive Program
- Command Verification Program
- Command Composition Program
- Spacecraft Status Display Program
- Science Instrumentation Performance Monitoring Program

b) Functional Description. The program will accept science and video command console inputs, run conflict checks to assure proper code input format, compose individual and blocks of science and video command message in NASCOM compatible format, time-tag command messages, maintain updated command lists of all science and video commands and their ordered sequence of execution, establish proper priority sequence as new commands are composed, and decode science and video command messages for checks against command request.

c) Interface Definition

- Program Inputs. Program input will be in the form of cards, tape, and/or disk; console inputs; prestored science and video command sequence.
- Program Outputs. NASCOM compatible time-tagged science and video command messages to the Command Composition Program; updated science and video command list; science and video commands failing conflict checks.

Command Composition Program

a) Related Programs

- SFOF Resident Executive
- Command Verification Program
- Spacecraft Status Display Program
- Science and Video Command Generation Program

b) Functional Description. The program will accept command console inputs, run conflict checks to assure proper code input format, compose individual and blocks of command message in NASCOM compatible format, time-tag command messages, maintain updated command lists of all commands and their ordered sequence of execution, establish proper priority sequence as new commands are composed, and decode command message for checks against command request.

c) Interface Definitions

- Program Inputs. Program input will be in the form of cards, tape, and/or disk; console inputs; prestored command sequences.

- Program Outputs. NASCOM compatible time-tagged command messages; updated command list; commands failing conflict checks.

Command Verification Program

a) Related Programs

- SFOF Resident Executive

- Command Composition Program

- Spacecraft Status Display Program

b) Functional Description. This program extracts command messages in NASCOM compatible format from a buffer system, checks command against a permissive command list, identifies command priorities, maintains a list of DSIF and SFOF commands transmitted, verifies the execution of DSIF transmitted commands, and maintains current status of command executions.

c) Interface Definition

- Program Input. Program input will be from the Command Composition Program via a computer buffer.

- Program Outputs. Command messages transmitted to DSIF; command messages transmitted by DSIF; tabulation of commands executed; commands failing verification checks.

Video Processing Program

a) Related Programs

- SFOF Resident Executive Program
- Video Enhancement Program
- Guidance and Control Subsystem Simulation Program

b) Functional Description. This program will receive the video data from the deep space stations via telemetry link and disk storage. Its primary function is to correlate video data with spacecraft position, attitude, and other parameters of interest. In addition, it formats data for further processing by the data enhancement program. Other functions are to:

- Assemble the data stream into complete frames
- Check for proper frame synchronization, continuous timing, and recognition of data dropout periods
- Calibrate the data
- Select the status words from the data stream
- Monitor the status of the video equipment and issue alarm messages if required
- Produce a binary tape containing digital video data for input to the Video Enhancement Program.

A printout will be produced for each video picture detailing the spacecraft position and attitude as well as other data pertinent to analysis of the video.

c) Interface Definitions

- Program Inputs

  Video data from disk storage

  Spacecraft position and attitude data from flight-path analyses and the Guidance and Control Subsystem Simulation Program

- Program Outputs

  Binary tape containing digital video data

  Printouts of spacecraft position and attitude and other functions necessary for video analysis

  Spacecraft video parameters to be monitored.

Video Enhancement Program

a) Related Programs

- Video Processing Program

b) Functional Description. This program edits, correlates, and filters digitized video data for the purpose of resolving bad spots, improving object definition, and enhancing picture quality through optimization of the development process.

c) Interface Definition

- Program Inputs. The program input will be in the form of punched cards and magnetic tapes with disk files used for intermediate storage.

- Card Input. Run parameters, output control data, table alterations, and execution data will be input by means of punched cards.

- Tape Input. Raw or preprocessed video data will be input from magnetic tapes and stored on disks for the processing cycle.

- Program Output

  Tabulated output:

  All card input including source and output identification will be recorded in tabulated form.

  Tape output:

  A video data output tape will be generated containing all data processed by the program along with those portions of unprocessed data selected by card input. This tape will be compatible for input to the video display device.

Maneuver Analysis and Command Program

a) Related Programs

- SFOF Resident Executive

- S-Band Radio Subsystem Simulation

- Spacecraft Status Display Program

- Guidance and Control Simulation Program
- Propulsion Module Simulation Program

b) Functional Description. This program is capable of evaluating operational requirements and constraints imposed upon the flight trajectories so that the effects of projected vehicle maneuvers may be accurately predicted. It is the primary off-line in-flight and preflight trajectory planning tool available to SFOF personnel. This program provides the basis for examining and implementing various maneuver policies and operational sequences that will best utilize vehicle capabilities. In addition, this program will aid in maneuver success evaluation activities.

The Maneuver Analysis and Command Program is divided into functional modules as shown in Figure 7-3. Under the control of a program executive, each module can be executed independently or as needed in predetermined computational sequences. This program will be able to simulate all maneuvers and mission trajectory phases subsequent to planetary vehicle trans-Mars injection to evaluate the generation of spacecraft maneuver aiming points. Typical program modules will include:

- The Executive Monitor. Sequences the computational modules dependent upon the program input. After all the computations are performed, the executive monitor takes command and awaits further inputs. The executive monitor logic also controls, by input, the parameters and information printed out.

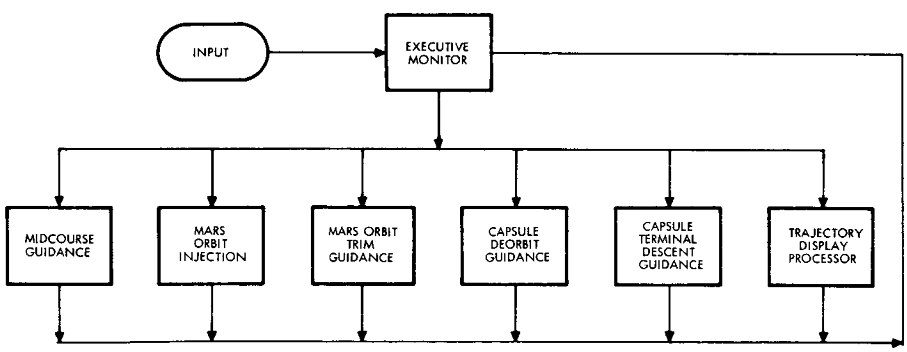


Figure 7-3
MANEUVER ANALYSIS AND COMMAND PROGRAM interfaces with all mission phase modules.

- Initial Estimator. Computes analytic values of all parameters required to initialize the precision iteration procedures

- Midcourse Guidance. Computes the "optimum" midcourse velocity correction which will ensure appropriate spacecraft Mars encounter conditions. This module computes the uncertainties in the execution of the midcourse velocity correction and transforms these into uncertainties in the terminal parameters. This module also computes the required roll and pitch/yaw commands required for thrust axis alignment.

- Mars Orbit Injection. Computes the appropriate guidance and control system parameters required to implement the spacecraft Mars orbit injection maneuver

- Mars Orbit Trim Guidance. Computes the required guidance and control system parameters necessary to implement the spacecraft Mars orbit trim maneuvers

- Capsule De-Orbit Guidance. Computes the required capsule guidance and control system parameters necessary to implement the capsule de-orbit maneuver

- Capuse Terminal Descent Guidance. Computes the required capsule guidance and control system parameters to assure capsule soft landing at the designated Mars landing site

- Trajectory Display Processor. Computes trajectory planning data such as sequence of mission trajectory events, trajectory geometry information, tracking station visibility, and trajectory-dependent photographic parameters.

c) Interface Definition

- Program Inputs

  Trajectory state vector information

  Reference epoch

  Orientation data

  Propulsion module parameters

  Estimated aim point information for each maneuver

  Landing site information

- Program Outputs

  Approximate fuel budget

  Trajectory display parameters

  Maneuver command data

  Maneuver uncertainties.

7.1.3 SFOF Simulation and Training Program

a) Related Programs

- SFOF Executive Program
- Telemetry Processing Program
- Spacecraft Response Simulation Program

b) Functional Description. This program will generate a simulated spacecraft telemetry data stream and can be employed in two modes of operation. In the first mode, the program will supply representative telemetry data for SFOF on-line real-time software system qualification testing. In the second mode of operation, the program will supply telemetry data for preflight operations planning, personnel training, and operational readiness testing for standard and nonstandard spacecraft operations. In addition, simulation tapes will be generated from inputs from the Spacecraft Response Simulation Program.

c) Interface Definition

- Program Inputs. The SFOF Simulation and Training Program input will be in the form of magnetic tape and punched cards.

  Predetermined values for the various telemetered measurements will be on magnetic tape. However, if more practical, these data may be on cards.

  Certain parameters identifying and controlling the input data will be entered via punched cards.

  During a run, certain abnormal events may be interjected via keyboard or switches.

- Program Outputs. The SFOF Simulation and Training Program output will be off-line printer, magnetic tape, and disk.

The printer will indicate the times at which deviations entered through the keyboard have been accepted by the program.

The generated data will be written on tape for storage and for later entry into the SFOF Telemetry Processing Program.

7.1.4 Spacecraft Response Simulation Program

a) Related Programs

- SFOF Executive Program
- SFOF Simulation and Training Program

b) Functional Description. This program represents a significant advancement in capability for in-flight evaluation of a spacecraft's performance and preflight analysis and training over any program in existence. The program will be employed by spacecraft performance evaluation teams to isolate causes of nonstandard spacecraft operations, compare the effects of alternative operation sequences, and assist in the formulation of contingency plans. The program will simulate the response of the spacecraft to telemetered commands and/or on-board subsystem failures. In conjunction with the SFOF Simulation and Training Program, this program will provide simulated telemetry data for in-flight evaluation of flight operations, verification of subsystem functions and their interactions, and spacecraft system analysis and performance evaluation.

c) Interface Definition

- Program Inputs

  Simulated spacecraft command messages

  Simulated spacecraft malfunction behavior data

- Program Output

  Data required to generate a simulation tape via the SFOF Simulation and Training Program

  Tabulation of command, malfunction, and response events.

7.2 SOFTWARE UTILIZATION

The software system at HOSC, like the system at SFOF, can be subdivided into four categories:

- HOSC on-line, real time system
- HOSC simulation and support system
- HOSC simulation and training system
- Spacecraft Response Simulation Program.

Each of these categories is nearly identical with its counterpart at SFOF. The primary difference is due to the fact that commands cannot be transmitted to the DSIF or the spacecraft from HOSC. The other notable difference between SFOF and HOSC is the inclusion of an Executive Program at HOSC (see Figures 7-4 and 7-5).

7.2.1 HOSC Executive Program

a) Related Programs

- Telemetry Processing Program
- Video Processing Program
- Spacecraft Status Display Program
- Electric Power Subsystem Monitoring Program
- Thermal Control Monitoring Program
- Propulsion Module Monitoring Program
- Guidance and Control Subsystem Monitoring Program

b) Functional Description. The HOSC Executive Program has complete control over the programs and routines contained within the HOSC real-time system. It determines which routines are to be executed, allocates storage space, and controls execution of the routines in a time-share environment. The HOSC Executive Program will be the master program and all other routines will be slave programs. It also identifies certain HOSC routine outputs as other program inputs and verifies interprogram data format compatibility. The Executive Program will monitor and control input and/or output data for all other programs.

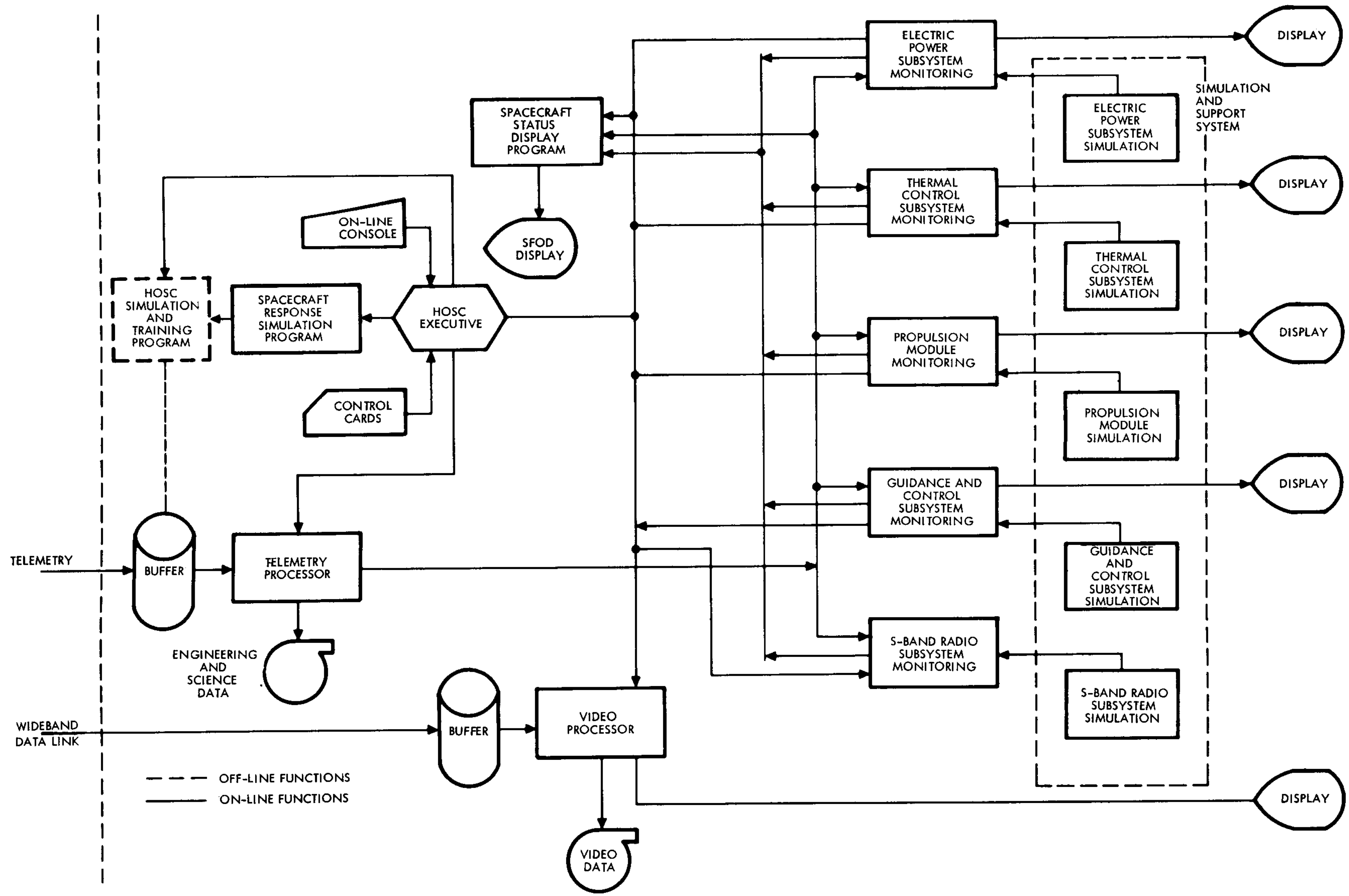


Figure 7-4
HOSC ON-LINE AND OFF-LINE SYSTEMS monitoring and simulation programs are functionally similar to SFOF systems except in command and science analysis capabilities.

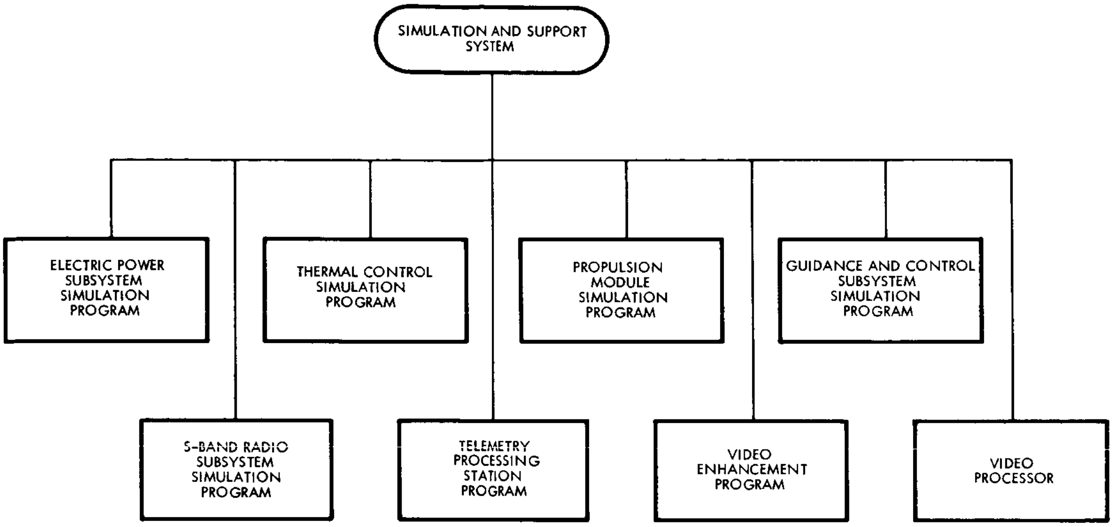


Figure 7-5
HOSC SIMULATION AND SUPPORT SYSTEM off-time computer program interfaces are like those at SFOF, but do not include maneuver analysis and command programs.

c) Interface Definitions. Inputs will be by cards and/or keyboard inputs and outputs will be on-line messages, cards, and/or tapes.

7.2.2 Comparison of HOSC/SFOF On-Line, Real-Time Systems

a) Related Programs

HOSC	SFOF
• HOSC Executive Program	• SFOF Resident Executive
• Telemetry Processing Program	• Telemetry Processing Program
• Electric Power Subsystem Monitoring Program	• Electric Power Subsystem Monitoring Program
• Thermal Control Subsystem Monitoring Program	• Thermal Control Subsystem Monitoring Program
• Propulsion Module Monitoring Program	• Propulsion Module Monitoring Program
• Guidance and Control Subsystem Monitoring Program	• Guidance and Control Subsystem Monitoring Program
• S-Band Radio Subsystem Monitoring Program	• S-Band Radio Subsystem Monitoring Program

HOSC	SFOF
• Spacecraft Status Display Program	• Spacecraft Status Display Program
	• Science Instrumentation Performance Monitoring Program
	• Science and Video Command Composition Program
	• Command Composition Program
	• Command Verification Program

b) Differences

- All the above monitoring programs at HOSC are directly slaved to the HOSC Executive Program. The similar programs at SFOF are slaved to the SFOF Resident Executive Program.
- Spacecraft Status Display Program: At SFOF, the spacecraft reference trajectory status is input from the Maneuver Analysis and Command Program, also at SFOF. At HOSC, the Spacecraft Status Display Program receives its trajectory information from SFOF.

7.2.3 Comparison of HOSC/SFOF Simulation and Support Systems

a) Related Programs

HOSC	SFOF
• Electric Power Subsystem Simulation Program	• Electric Power Subsystem Simulation Program
• Thermal Control Subsystem Simulation Program	• Thermal Control Subsystem Simulation Program
• Propulsion Module Simulation Program	• Propulsion Module Simulation Program
• Guidance and Control Subsystem Simulation Program	• Guidance and Control Subsystem Simulation Program

HOSC	SFOF
• S-Band Radio Subsystem Simulation Program	• S-Band Radio Subsystem Simulation Program
• Telemetry Processing Station Program	• Telemetry Processing Station Program
• Video Processing Program	• Video Processing Program
• Video Enhancement Program	• Video Enhancement Program
	• Maneuver Analysis and Command Program

b) Differences. The eight HOSC programs listed above are identical to their counterparts at SFOF. The one difference is that the Video Processing Program is slaved to the HOSC Executive Program at HOSC, and is slaved to the SFOF Resident Executive at SFOF.

7.2.4 Other HOSC Programs

HOSC Simulation and Training System and the Spacecraft Response Simulation Program. The third and fourth categories of the HOSC software system are identical to the systems at SFOF with the exception that they will be slaved to the HOSC Executive Program.

7.3 DSIF TCP COMPUTER PROGRAM

This specification describes the computer program operation in the planetary vehicle telemetry and command processor (TCP) at the Deep Space Instrumentation Facility.

7.3.1 Functional Description

The DSIF PV-TCP program monitors two activities — telemetry data handling and commanding functions. The program primarily serves as a link between the planetary vehicle and the SFOF and HOSC for real-time handling of telemetry data and command functions. Basic DSS/TCP functions include acquisition of Voyager PCM data; monitoring PCM reception error rate; driving local status displays; providing serial undecommutated data to communication interfaces; and formating commands received from SFOF for transmission and monitoring command transmission on a bit-by-bit basis.

More substantial PV-TCP involvement occurs during backup activities when the SFOF facilities are relieved for portions of the cruise phase and certain other noncritical mission phases. Backup operations are also performed during failure modes caused by inability to communicate with the SFOF through standard links or by unavailability of the SFOF computer systems. The PV-TCP program will operate in the backup mode during certain test mode operations.

7.3.2 Telemetry Data Handling

The PV-TCP program, in conjunction with other DSIF voyager mission-dependent equipment, has the capability of monitoring the low-rate and the high-rate data links simultaneously. The level of telemetry handling is significantly different for the two.

a) Low-Rate Data Link. The low-rate telemetry data link primarily contains engineering and status information of immediate concern. In addition, the reception rate is well within the tolerance of the DSIF MDE. The bulk of the telemetry data handling functions are, therefore, concerned with this link (Figure 7-6). Such functions include:

- Determining word and frame synchronization of the data stream. The synchronization algorithm is based

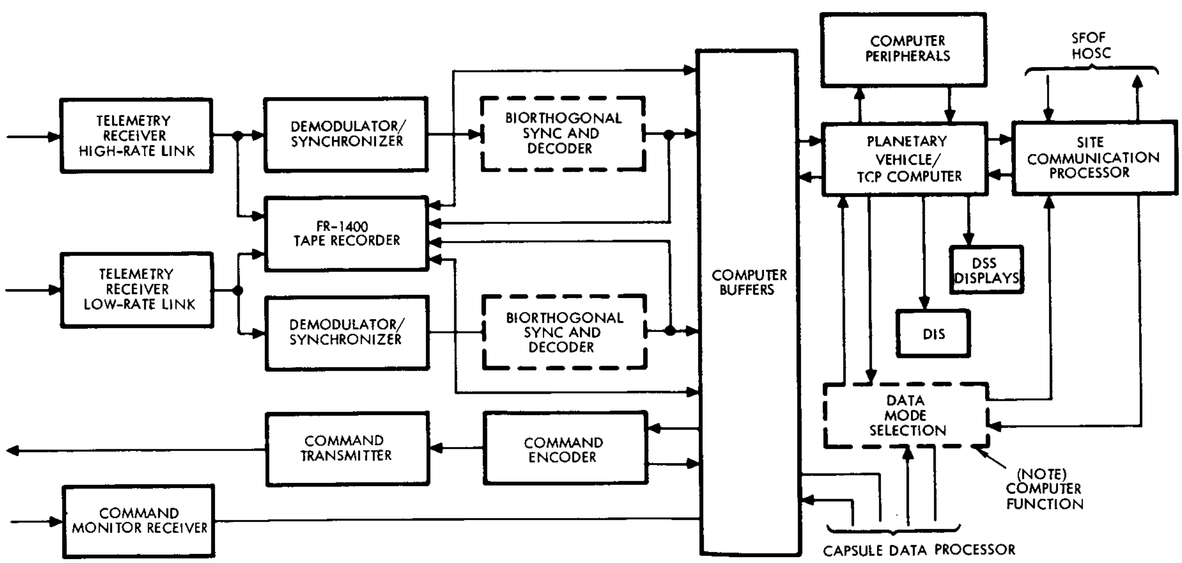


Figure 7-6
DSIF TCP COMPUTER INTERFACES with the MDE and MIE at the DSS as shown, with the data flow within the system. Dotted lines indicate functions which are accomplished by a combination of hardware and software implementation.

on three synchronization words in a main frame composed of 64 six-bit engineering channels or 96 six-bit engineering and low-rate science channels. When synchronization is possible, an indicator is included with the data transmitted to SFOF and HOSC. Otherwise, the raw data stream will be transmitted in blocks for more complete processing at SFOF and HOSC.

- Determining telemetry error rates considering system performance criteria and thresholds. Errors are detected and corrected to the extent possible with the capabilities of the MDE. Error rates are constantly monitored and reported.

- Decommutating the data and selectively editing particular vehicle parameters, status indicators, and other information of immediate interest

- Analyzing mode identification indicators as to bit rate, format, and data source of the high-rate data link in order to determine the processing and routing requirements of the data received on that link

- Driving local displays with selected planetary vehicle parameters and generating alarm signals when appropriate

- Formating, time-tagging, and delivering blocks of telemetry data to the site communications processor for NACOM-compatible processing and subsequent routing to SFOF and HOSC. Recording the raw data stream on digital magnetic tape.

b) High-Rate Data Link. The high-rate data link may contain data from a number of vehicle sources. The major function of the PV-TCP in this instance is to route the data as appropriate for further processing, e.g., capsule video to the capsule contractor TCP, science data to SFOF and HOSC, etc. The data rates are such that a mid-1970 computer capability is assumed at the PV-TCP. In addition to routing, data handling functions include recording on digital tape, and reversing inverted data transmissions.

7.3.3 Command Data Handling

a) Standard Operation. The PV-TCP program performs the following standard command functions:

- Formating command instructions received from SFOF, verifying the format, and transmitting commands to the command encoder at the appropriate times

- Verifying command encoder transmission by performing a bit-by-bit check and signaling the command encoder to interrupt transmission in the event of an error.

b) Nonstandard Operations. Manual origination of commands at the command generator and limited verification of command execution by examination of telemetry status bits and telemetry analysis will be performed.

7.3.4 Interface Definition

The DSIF PV-TCP computer program interfaces with DSS devices via the computer buffer.

a) Program Input

- DSIF MDE Input. The computer buffer will generate interrupts to the program from the following mission-dependent equipment:

  Command encoder
  Demodulator-synchronizer
  Bi-orthogonal synchronizer and decoder.

- DSIF MIE Input. The computer buffer will generate interrupts to the program from the following mission-independent equipment:

  Monitor receiver
  Site communications processor
  Time code generator and/or the millisecond clock.

b) • Program Output. The DSIF PV-TCP will control the operation of the following DSS devices:

  Command encoder
  Local displays
  Selected linkages with the site communications processor.

8. PROGRAMS

8.1 COMPUTER PROGRAM DESIGN

Within the framework of the milestone product concept, the design of spacecraft contractor-furnished computer programs in support of Project Voyager will be directed towards the achievement of milestone 1. Milestone 1 is a general program requirements document which:

- Identifies available data handling support characteristics, capabilities, and limitations
- Specifies the requirements each computer program must satisfy to provide the computing support necessary for Voyager preflight and inflight mission operations, and
- Provides information for timely program development, field integration, and acceptance by the customer.

As noted in the previous section, the spacecraft contractor will furnish computer software systems at the Deep Space Instrumentation Facility, the Space Flight Operations Facility in Pasadena, and the Huntsville Operations Support Center. The three operational centers will have different data handling equipment, standard practices, and resident software support capabilities. The programming design effort will be quite complex because of:

- The different capabilities of each operational site
- The desire to design software systems which will permit evolutionary development for later generation of Voyager mission support, and
- The ambitious aim to develop a sophisticated simulation program system for operational readiness testing, personnel training, in-flight system performance monitoring, and in-flight diagnosis of faulty system operation.

The successful development and implementation of the simulation program system will constitute a major advance in the state of the art. Current telemetry simulation programs have the capability to generate tapes for standard spacecraft system operation and a limited number of nonstandard operations. The proposed simulation system will provide telemetry data corresponding to standard, and a wider class of nonstandard operations will reflect spacecraft response to command activity and will account for the appropriate interaction of one subsystem upon another.

This program design effort will entail software requirements definition, design tradeoff analyses, and individual program requirements definition. Data handling equipment, software interface, and procedural requirements will be established for each site. System-level functional requirements will be defined. Program design criteria will be developed, data flow analyses will be performed, and individual program functions will be identified. To implement program functions, algorithms will be selected and the logic flow will be designed. Program design feasibility will be established and the result of design tradeoff analyses will be documented. Detailed design, interface, development, delivery, and acceptance requirements will be formalized by the milestone 1 program requirements document.

8.1.1 Software Requirements Definition

The purpose of this software design activity is to develop a sequence of baseline requirements for the mission-dependent computer programs to be furnished by the spacecraft contractor for DSIF, SFOF, and HOSC.

Data handling equipment and resident software interface requirements will be identified for each of the operational sites. Software requirements reflecting established system procedures at these sites will be defined. Functional software requirements will be developed to provide mission operations personnel with the necessary computing support.

This effort will require considerable liaison between mission operations personnel and personnel familiar with the capabilities of the various operational sites. If the software systems to be implemented at each site are to have maximum effectiveness, the design team will have to be thoroughly conversant with the mission operations procedures planning effort.

8.1.2 Functional Requirements

It will be necessary to establish the functional requirements that the spacecraft contractor-furnished software must satisfy at each operational center. A sequence of functional requirement baselines will be established so that design feasibility can be evaluated as early as possible.

The development of the functional requirement baselines will depend on considerable consultation with cognizant mission operations personnel who will be engaged in spacecraft subsystem monitoring, simulation and training, in-flight diagnostic evaluation, trajectory planning, data correlation, and recording activities prior to and during flight operations. It is essential that these liaison and consultation activities be initiated as soon as possible and continue until program field integration and acceptance. It should also be recognized that the operations programming design effort could influence substantially the mission operations procedures that will be developed.

The spacecraft system monitoring activity during flight operations is a complex one because of the long duration of the mission, the large number of telemetered system performance parameters, the varied data rates during high and low mission activity phases, and the simultaneous operation of two spacecraft. Desired and required monitoring and display capabilities will be identified during consultation with mission operations personnel responsible for each individual subsystem.

Effective training of operations personnel, extensive software qualification testing, and comprehensive operational readiness testing will impose demands for highly sophisticated telemetry simulation programs. As previously noted, it would be very desirable to design simulation programs which will generate telemetry data simulating the spacecraft system response to commanding activity. With such a capability, it will be possible to effectively train personnel to identify nonstandard spacecraft operations, isolate causes of faulty system operation, prepare and execute corrective commands, and observe the simulated response of spacecraft systems to command activity in a test environment.

If spacecraft subsystems are not functioning properly, SFOF flight operations personnel must have some capability to analyze the cause of faulty operation and initiate command activity that will result in a return to standard operations. In the event the latter is not possible, it will be necessary to determine the best permissible mode of spacecraft operation. It will be difficult to isolate the cause of faulty subsystem operation because the diagnostic analysis must be based on telemetered subsystem performance parameters that may be highly dependent on each other. Quick-response diagnosis of faulty operation and timely execution of the appropriate command sequence will require considerable computing support. The design of computer programs to aid this effort could influence the choice of parameters to be telemetered back to earth. Modeling of subsystem operation for simulative interaction between one subsystem and another will be necessary. As noted, the simulation of telemetry signals reflecting spacecraft response to command activity will necessitate a major design effort.

The composition, coordination, accounting, verification, and execution of spacecraft commands will also require considerable computing support. The command status accounting procedures and the criteria for establishing command sequence priorities will require considerable attention.

Because the spacecraft real-time trajectory planning effort will depend on the capsule soft-landing requirement, a sophisticated spacecraft maneuver analysis and automated command composition capability will be needed. There is a need to display a multitude of trajectory information for a larger number of critical trajectory events such as maneuvers, earth occultation, spacecraft eclipse, and photographic site overpass conditions. It will be necessary to define the parameters characterizing each mission trajectory-dependent event and the manner in which these parameters will be displayed.

Mission operations personnel will desire the capability to correlate information for each of the subsystems and relate this information to current and planned trajectory events, command activity schedules, science and video data, and possible observatory data concerning solar flare activity. Such data correlation capabilities will necessitate the development of mission procedures heavily dependent on computing support.

a) Data Handling Equipment Interface Requirements. For each operational center software/equipment interface requirements will be established. The data handling equipment at the SFOF will be mission-independent or furnished by the capsule and spacecraft contractors.

The characteristics, capabilities, and limitations of computer processor units, computer buffer systems, computer peripheral systems, data transmission links, recording devices, and display equipment will be identified. Storage capabilities, timing restrictions, input/output data format, and operating mode requirements will be defined.

b) Resident Software Interface Requirements. Available at each of the operational centers will be data processing software with which the spacecraft contractor-furnished software must interface. Such resident software will include status display drivers, communication processor software, executive programs that control mission-dependent and -independent software, and tracking data analysis and orbit determination programs.

Requirements relating to the interfaces with this resident software will be determined for each operational site. Such interface requirements will relate to program overlay configurations, time sharing, program interrupts, input/output formats, buffer storage, and timing restrictions.

c) Site Procedural Requirements. DSIF, SFOF, and HOSC procedural constraints affecting the design of the computer program systems will be defined. Such procedural constraints will reflect video data handling, tracking data handling, capsule data handling, and general purpose display data handling requirements. It will also be necessary to delineate the existing standard procedures and practices that have been developed at each site.

Additional effort will be required to define the mission-dependent and -independent software integration procedures and the software configuration control practices necessary to assure orderly and efficient design, development, and maintenance of operations software. These requirements will influence the software system design because of the need to plan for cost-effective accommodation of post-development design changes and future software capability upgrading essential to a Voyager program time span of long duration.

8.1.3 Design Tradeoff Analyses

The purpose of this design activity is to develop the functional design requirements for each individual computer program to be furnished at each operational site. The design requirements for each computer program will be developed from the software system requirements baseline available for this purpose.

Program design criteria will be identified, a complete system data flow analysis will be performed, individual program functions will be specified, algorithms and logic will be developed for each assigned program function, and program design feasibility will be established. Design tradeoff analyses will be performed to justify selection of the design approach taken.

a) Design Criteria Development. Design criteria will be developed which, together with the available design requirements baseline, will enable selection of the best design approach from among as many design alternatives as may be identified. These design criteria will serve to ensure practical, cost-effective, and flexible operations software development, implementation, and maintenance. The design criteria will be such as to allow evolutionary software capability upgrading so essential to satisfy increasing computing support requirements for the later generation Voyager missions.

b) Data Flow Analysis. The purpose of this design effort will be to establish the functional design requirements for each computer program within the appropriate software system. Preliminary program interfaces will be delineated to assure software and data flow compatibility.

The characteristics of the serial pulse code modulated data stream will be defined. Also, the software system output requirements will be delineated to a finer level of detail. Low-rate engineering, low-rate science, medium-rate science, and high-rate video telemetry data flow characteristics will be analyzed in detail. The software system outputs will provide a complete definition and format of the parameters to be recorded, displayed, printed, plotted, and buffered. The simulation predict parameters, alarm signals, and processed video picture characteristics will be identified. The manner in which these parameters are presented to the user and the required equipment interfaces will be specified.

The data processing functions required to derive the software system outputs from the incoming serial bit stream will be defined for each operational site. It will be necessary to define where these functions are to be performed, to determine which programs in the software system will perform the functions, and to establish the compatibility of program interfaces.

The following telemetry data processing functions have been identified to date: data acquisition from a computer buffer, telemetry error rate determination, display driving, data output buffering, word and frame synchronization, decommutation, selective and/or complete editing, data mode and source identification, alarm processing, recording, decoding, routing, reversal of inverted data transmission, frame assembly, data calibration, frame synchronization checks, vehicle clock to GMT conversion, data unit conversion, status change printing, and video reconstruction and enhancement.

The following command data processing functions have been identified to date: command message composition, format checking, conflict checking, permissibility checking, time tagging, command status accounting, command priority identification, command list maintenance, command execution verification, command instruction formating, and command transmission verification.

The software system requirements baseline and the program design criteria will serve as guidelines to ensure that the proper data is extracted from the incoming data stream, routed to the appropriate user area, and processed for presentation to analysis teams in the specified manner. The computer program functions will be specified after the appropriate design tradeoff analyses are performed. Computer program input/output parameters and their formats will be defined. The simulation and support computer program functions will also be specified.

c) Algorithm Selection and Logic Design. After each program function, input, and output has been defined, it is necessary to design the appropriate algorithms or logic by means of which the program output will be derived from the input. Often alternative techniques must be investigated to assure the best means of implementing the program functions.

Such techniques will include modeling of subsystems for simulation predict determination, trajectory simulation procedures, maneuver analysis computations, alarm signal processing, decoding algorithms, and command status accounting procedures. Executive monitor requirements for the Maneuver Analysis and Command Program will have to be developed.

Software systems executive logic requirements will be established to assure the most effective on-line and/or off-line use of the simulation and support programming system prior to or during flight operations.

d) Feasibility Evaluation. The feasibility of the software system design will be evaluated against pre-established software system requirements and design criteria. All interface, procedural, and functional requirements will be shown to be satisfied.

Special emphasis will be given to identification of those requirements that will limit the evolutionary growth of software capabilities. Future design effort may be directed towards the upgrading of appropriate equipment capabilities, the improvement of existing standard site procedures, or the planning of more effective operations procedures.

8.1.4 Program Requirements Definition

The purpose of this activity is to document the design, interface, development, and acceptance requirements for each individual computer program to be field-integrated. Program functions will be described together with the algorithms and/or logic required by each function. All program interfaces with data handling equipment, contractor-furnished computer programs, and resident site software will be described. In addition, the program operating requirements and guidelines will be specified. The completion of this effort will constitute achievement of milestone 1 within the milestone product schedule.

a) Functional Description. For each computer program, statements describing all program functions will be provided. This functional description will be supplemented by the necessary logic flow, equations, and/or algorithms required to implement each program function. The level of detail will be such as to provide the necessary information for the development of detailed computer program specifications.

b) Program Interface Requirements. The characteristics, capabilities, and limitations of the data handling facilities and equipment with which the program interfaces will be described in detail. Input variables, constants, and the formats of each will be specified. Output variables, formats, and media will be identified. Interfacing computer programs will also be identified.

c) Program Operating Requirements. Operating mode requirements will be identified for each program. Such operating modes will specify whether a) the program will be under executive control, supporting on-line functions; b) it is to be used off-line; and c) it is to be used exclusively for operational readiness testing, personnel training, or flight operations support. All program control, interrupt criteria, call-up and halt and recovery requirements will be identified. Site procedural limitations and/or requirements will be specified.

d) Program Delivery and Acceptance Requirements. Schedules for program development, delivery, integration, and acceptance tests will be provided. These schedules will be coordinated between the project office and the computer programming agency in a manner which is acceptable to each organization.

   Program field integration requirements and acceptance test criteria will be provided to ensure that these activities will proceed smoothly. The DSIF-TCP program will be tested at TRW by simulating all program/DSIF equipment interfaces prior to delivery.

8.2 PROGRAM DEVELOPMENT AND INTEGRATION

The development and integration process for Voyager mission-dependent computer software, within the framework of the milestone product concept, commences in response to the program requirements document (milestone 1). A functional flow diagram of these activities is shown in Figure 8-1.

8.2.1 Program Generation Request

As a result of mission requirements definition, software requirements definition, and system characteristics definition, the program requirements document (milestone 1) is generated. Accompanying the Milestone 1 document will be a program generation request. This serves as the formal request from the project office for the generation of a specific element of computer software whose requirements are defined in milestone 1. In response to the request, a cognizant lead programmer/analyst is designated for each program to be responsible for its development and integration.

8.2.2 Program Specification Development

In response to the program generation request and milestone 1, the cognizant programmer will prepare a preliminary draft of the program specification based upon the information furnished him prior to the start of coding. During the preparation of the rough draft, he will maintain close technical contact with the cognizant engineer to resolve directly any conflicts that may arise. This document will serve as a technical statement from the programmer indicating the manner in which the program request will be translated into an appropriate computer program. The rough draft of this document will:

a) State what the program must do and how it must function in relation to an input-output device, other programs, and Voyager mission requirements

b) State specifically the relationship of this program to other programs regarding input-output, tables of data, control information to-from programs, and input-output devices

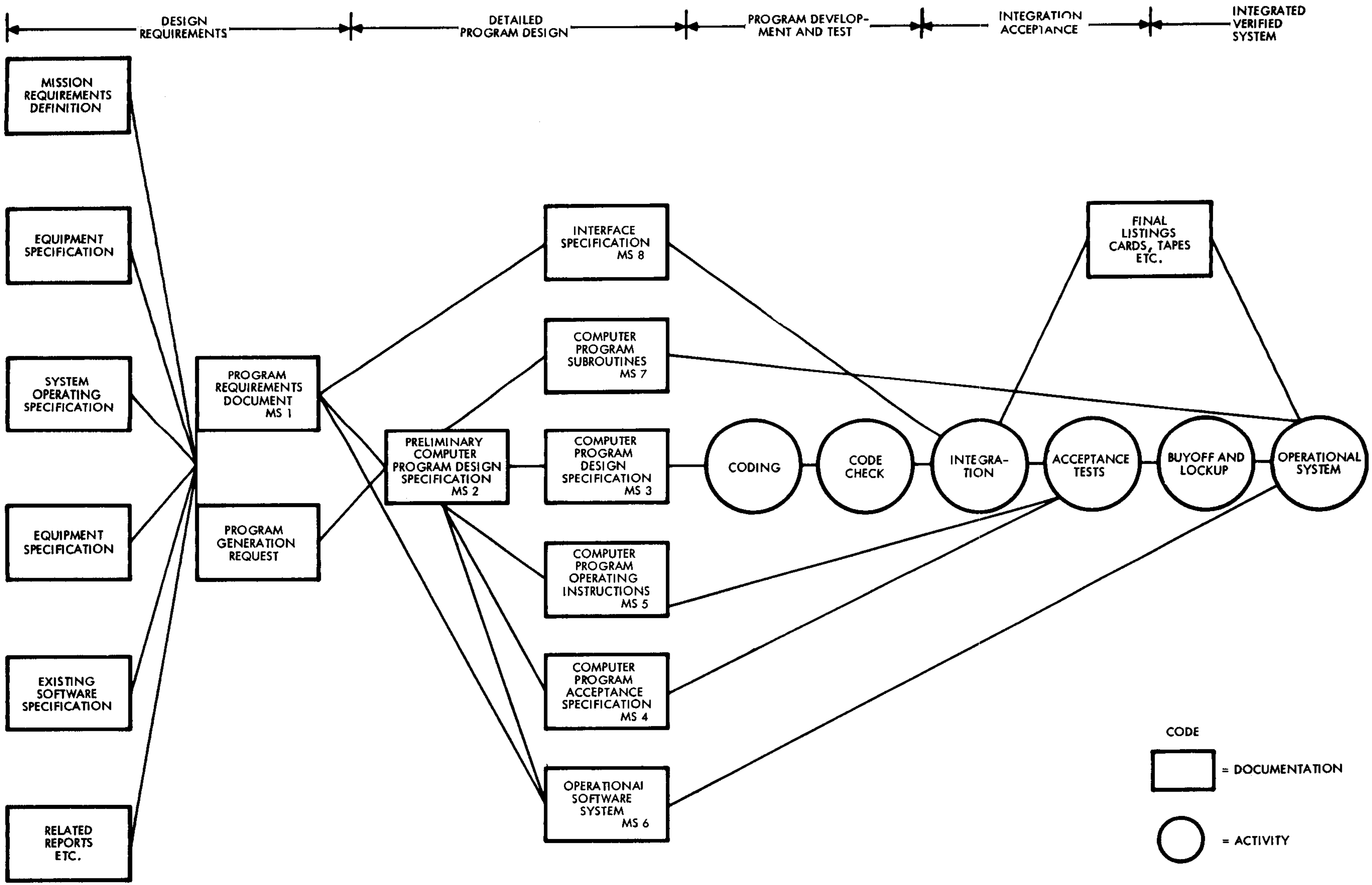


Figure 8-1
DEVELOPMENT AND INTEGRATION OF SOFTWARE is carried out as illustrated in this flow activity chart.

c) Estimate timing restrictions such as minimum time the program must not be interrupted by another program

d) Show flow charts and describe the logical interaction of various subroutines, input-output devices, and portions of the program which do not necessarily fall into this category (executive routines). In cases where a subroutine is not appropriate, the functional operation of the program will be indicated.

e) State the method by which the program will be checked; describe the program drivers that have to be coded; discuss any limiting factors of the checkout process; estimate the computer time required for checkout; and provide a test plan.

When published in final form, the computer program design specification will contain detailed program listings and program descriptions which will:

a) Identify the program by title, deck, or tape number; author and date; machine, configuration, and source language; and DSN/HOSC functional area

b) State the purpose of the program

c) Define all restrictions on its use such as components or programs required, data quantity, data form, and critical timing

d) Prescribe the usage including calling sequence, storage space required, error codes and messages, and format received or generated

e) Specify coding information including constants and their locations and erasable input-output locations

f) Describe the checkout status and method

g) State the required program execution time for representative computations

h) Specify all tables by name, type, function, size, indexing, structure, and program usage.

All program listings will be accompanied by sufficient commands which establish the relationship between various steps in the detailed flow charts and the program code.

8.2.3 Coding/Code Checking

The coding of the computer programs is accomplished in two steps. First, the routines are coded and checked after appropriate analyses of engineering algorithms have qualified the performance of the formulations. Following this, the complete program is coded by integrating the routines and then checking against expected program operation results for typical computational sequences.

The checkout (code checking) of the programmed equations consist of tests designed to verify the coding process and to ensure that all branching, output, decision criteria, and phase termination logic have been correctly mechanized. Testing of the program is accomplished to qualify the programming and the overall integrity of the software logic by exercising the program with the appropriate data profiles and noting the program responses. Test evaluation consists of detailed, direct comparisons of programmed responses with the specification and precalculated expectations.

8.2.4 Program Integration

Upon completion of the design and development phases of the software implementation cycle, the configured computer program enters the integration phase. During the integration phase, all program elements and subelements are integrated, in a tree-structure form, into a composite computer system program. The types of software elements to be integrated, in order of complexity, are:

- System
- Subsystem
- Program
- Subprogram
- Routine
- Subroutine
- Tables
- Macro-instruction
- Test data.

After all program elements and subelements are integrated into the operating system program, the program will be exercised to ensure that all logical branching, output, input, decision criteria, and executive logic have been properly mechanized. During this integration/pretesting phase, all software routines will be thoroughly checked for operability, reliability, and fulfillment of design criteria specified in milestones 1, 2, and 3. This requirement is implicit in the software verification process due to complex interfaces with the real-time computer program and the necessity of time-sharing the computational activities in the main processor.

In general, each mission will be treated as a separate entity and the required software will be integrated and verified as individual packages to accomplish the end objective for the specific mission.

8.2.5 Program Documentation

The documentation involved with a software development is of two types. The first is that documentation which provides the necessary information for the development itself, and the second is the documentation generated during or as a result of the development.

The first categories are:

- Program requirements documents, milestone 1
- Preliminary computer program design specifications, milestone 2
- Interface specifications, milestone 8
- Equipment specifications
- Existing software specifications
- Related reports, publications, etc.
- Schedules
- Manpower allocations
- Software configuration management specifications
- Correspondence.

The second category includes:

- Computer program design specifications, milestone 3
- Computer program acceptance specifications, milestone 4
- Computer program operating instructions, milestone 5
- Operational software system, milestone 6
- Computer program subroutines, milestone 7
- Associated flow charts
- Program listings
- Correspondence
- Change notices
- Configuration control decisions
- Test results.

A major advantage of the milestone product development concept is that the development of computer program support documentation is integrated into the program development cycle in a time-phased relationship. The milestone documents 1 through 8 provide a comprehensive description of the program capabilities and operation.

Table 8-1 contains a list of sample documentation descriptors.

8.2.6 Computer Software Qualification

At the completion of the implementation process it is necessary to perform critical testing and, on the basis of these tests, to provide cerification that the software is ready for operational use. The testing, leading to certification, must be preceded by a planned series of preliminary, informal testing to increase the probability of certification and to cover the depth of testing appropriate to software for a "high-risk" mission.

Table 8-1. Sample Definition of Documentation Descriptors Used in the Mission-Dependent System Documentation

A. Document Identification
1. Title
2. Date
3. Author
4. Author's installation
5. Customer

B. Processor Element Identification
1. Element type
2. Title name
3. Title mnemonic
4. Number
5. Date
6. Source language
7. Author
8. Author's installation
9. Responsibility
10. Authorization
11. Related documents

C. Purpose
1. Task or function
2. Scope
3. Secondary applications

D. Method of Solution
1. Logical concepts
2. Mathematical equations and derivations
3. Mathematical or procedural solution
4. Decision flow diagram
5. Decision options

E. Equipment Requirements
1. Computer
2. Monitor or operating system
3. On-line units
4. Connection layout
5. Special requirements

F. Usage Requirements
1. Calling sequence and/or operating procedures
2. Arguments and/or parameters
3. Control card format and usage
4. On-line listings
5. Tape assignment and usage
6. Console operations
7. Start procedures
8. Restart procedures
9. Assignment of on-line units
10. Switch settings
11. Interrupt actions
12. Indications of processor flow
13. Retention and release schedule

G. Input Requirements
1. Media
2. Sources
3. Preparation
4. Description and parameters
5. Quantity and limits
6. Formats
7. Sample forms
8. Options
9. Interfaces

H. Output Requirements
1. Media
2. Recipient
3. Error indications
4. Description and parameters
5. Quantity and limits
6. Formats
7. Sample forms
8. Options
9. Interfaces

I. Restrictions
1. Accuracies
2. Error checks not made
3. Required execution of other processors
4. Reference and intercommunication pool (RIPOOL) requirements

J. Timing Requirements
1. Procedural
2. Input
3. Output

K. Storage Requirements
1. Computer core memory
2. Disc and/or tape storage

L. Interfaces
1. Input
2. Output
3. RIPOOL
4. Control
5. Utility
6. Next higher processor elements
7. Next lower processor elements

M. Special Compatibility Requirements

N. Updating Requirements
1. Types
2. Approval
3. Effectivity

O. Flow Diagrams

P. Sample Problem

Q. Validation
1. Specifications
2. Procedures
3. Test case

R. Acceptance Requirements
1. Procedures
2. Criteria
3. Historical records

S. Printing
1. Date
2. Distribution

T. References
1. Other programs
2. Bibliography

a) Acceptance Test Specification Development. Prior to the development of a milestone 5, computer program operating instructions, but after the issuance of the milestone 3, computer program design specification, the milestone 4, computer program acceptance specification, is developed.

The acceptance specifications are designed to provide detailed and systematic procedures for evaluating the computer program subsystem and for demonstrating that the subsystem meets the requirements of all applicable specifications. In addition, the specifications must sufficiently define conditions of acceptable performance so that the usefulness and workability of the subsystem may be reasonably determined.

In general, the tests are designed to:

- Test out and exercise the integrated software system, interfaces, coordination, and communication
- Test the functions and options at the concerned facility (DSN, HOSC) in a manner independent of other facilities
- Test the entire integrated system interface, coordination, and communication.

b) Acceptance Testing. Acceptance tests verify that separate elements of the Voyager mission-dependent software can perform together in accordance with the functional requirements specified, and that these requirements are compatible with the current space vehicle data configuration. Acceptance tests for mission-dependent software will be performed at two levels: program level and system level. Both will be subjected to critical operations level environment to verify performance characteristics and to identify potential failure areas.

During these tests, the mission-dependent software is exercised under a variety of conditions determined by combinations of operating modes, bit rates, command sequences, communications capabilities, and equipment configurations, which can occur during standard and specified nonstandard space operations. The tests also verify that all Voyager mission-dependent hardware and software interfaces are compatible by demonstrating acceptable program system functional performance. It will not be necessary that the tests be performed in real time, nor is rigid adherence to operational procedures required; however, the use of simulated space vehicle data is required. Successful accomplishment of these tests verifies that the software system is functionally capable of supporting the Voyager mission.

The computer program acceptance specifications formally defines a series of program acceptance tests whose successful completion is required for program certification and project release. It will be the responsibility of the technical analysis directors to direct its preparation using the program requirements document as a guide. This document will:

- Specify all program functions and options that have been designed into the program
- Identify all program data sources that will be used in standard or anticipated nonstandard program operation
- Describe all program output displays, both human and machine readable, which will be generated by the program in standard or anticipated nonstandard operation
- Define test evaluation criteria and program output acceptance standards. Certification will be based on the program's capability to meet such standards.

Analysis of computer program output data may be accomplished after the previously defined demonstration. This test will be witnessed by the data processing operations director, the technical analysis director, the cognizant programmer, and cognizant engineer. At each site (HOSC, DSN) the estimated duration of the test, the required personnel support, computing equipment, and test sequence of events will be provided. Data necessary for the conduct of the test will be specified and will reflect conditions encountered in anticipated standard and nonstandard Voyager space flight operations.

c) Buy-Off and Lockup. The successful completion of the acceptance tests, and the signing of the test record by all representatives, constitute the buy-off.

In the case of the SFOF, buy-off occurs when the program has become operational and has been entered into the SFOF data base. In addition, all support documentation has been completed, received, and distributed. In like manner, software buy-off activities for DSIF and HOSC will be conducted.

Lockup occurs when the operational program package is submitted to the site inventory control engineer (ICE) for approval and further action. Any further work on locked-up programs and support documentation is formally controlled through change control request procedures.

9. EOSE INTRODUCTION

The Voyager EOSE preliminary requirements and the integrated systems approach embodied in its design are discussed in Section 10, Test Philosophy.

The EOSE systems test complex (STC) endorses the concept of designing support equipment from the top down. Section 10, therefore, describes the STC in its full systems context from the standpoint of fulfilling all of the requirements necessary for the integration, test, and launch checkout of the complex Voyager spacecraft, including the requirements associated with integration of the capsule, payload science, and mating interfaces.

The STC results not only from a requirements analysis ensuring precise alignment of EOSE capabilities with the spacecraft requirements but, in addition, from the bank of experience which TRW has accumulated through the integration and test of other sophisticated space hardware systems in the past.

The system test complex encounters the spacecraft hardware after completion of the factory test program for each individual assembly and/or subsystem and, beginning at that point, provides the total capability necessary for the integration, system test, and launch checkout leading to readiness for liftoff at ETR.

EOSE is the prime electrical support system for the spacecraft and total planetary vehicle through verification, checkout, assembly, test, and prelaunch activities. It serves as the stimulus for spacecraft operations and as the depository of data generated by those operations. It is the source of commands and the recipient of spacecraft response. In its dual role of interrogator and verifier, the EOSE provides the means for determining the functional compliance of the spacecraft, recording a history of spacecraft operations, and displaying in real time the current status of the particular test sequence being experienced.

10. TEST PHILOSOPHY

The Systems Test Complex (STC) is the basic EOSE constituent. It is designed and operated to perform the complete closed-loop spacecraft functional evaluation program. Although the STC is configured to automatically cycle through the complete spacecraft integrated systems test (IST), it is designed for manual override of any portion of the IST and for the complete manual operation of the system when necessary. The ability to manually control is an essential function for fault isolation and special or unusual tests. Whether in the automatic or manual mode of operation, the computer receives and analyzes all data and stores periodic samples of data which are coded by time and unit for future reference in the form of history tapes. History tape readout ability allows for trend evaluation and long-term degradation analysis.

The computer software is modularized such that specific portions of the IST have separate routines which are available in an integrated mode or are usable individually. This form of programming provides a flexibility for modifying, replacing, or adding a specific test routine.

The STC is configured to communicate with the spacecraft through normal telemetry and command channels and to functionally check the operating parameters that are to be evaluated during the spacecraft mission. However, there are engineering data which are not available through telemetry data transmission; there are operating tolerances which are more precise than telemetry data can provide; and therefore, there are requirements for continuous sampling of dynamic analog measurements. In these limited cases a hardline interface to the spacecraft will be utilized.

The STC will be used in unmodified form for all test locations including the launch site. This is advantageous from the standpoints of maintaining the data bank, maintaining validated interfaces, and test crew familiarity with the equipment.

11. EOSE OBJECTIVES AND DESIGN CRITERIA

11.1 OBJECTIVES

Basically and in the most general sense, the objective of the EOSE is to provide the capability for testing the spacecraft to the depth required to develop the desired level of confidence in the spacecraft performance. The more specific objectives are:

- Monitoring. The EOSE must provide continuous monitoring of all spacecraft subsystems. With continuous monitoring, subtle design flaws and transients are less likely to escape detection; it also becomes easier to establish trends and detect incipient failures. Some critical spacecraft parameters, such as battery voltage, current, and platform temperatures, will be displayed continuously for monitoring by test personnel. In addition, the STC computer will compare all spacecraft parameters with predetermined limits and will alert test personnel to any out-of-tolerance conditions.

- Enhancement of Personnel Efficiency. The EOSE will provide as much automatic parameter checking and data processing as possible to reduce the number of personnel needed to perform tests. This decreases both the cost and confusion associated with major system tests. Most spacecraft data and all critical parameters are entered into the STC computer which will alert test personnel to anomalous test conditions.

- Test Time Efficiency. The EOSE will reduce the amount of time required to perform major system tests. This not only decreases the cost of a test but reduces fatigue in test personnel which is a major cause of human error. The capability of the STC to perform many functions automatically and to process large amounts of data quickly renders the whole test process more efficient. For example, command sequences can be executed much more quickly and accurately by the STC computer than by an operator manually selecting and executing command sequences. Stimulus equipment can be set up automatically by the STC computer, deriving the same benefits in expediting tests and avoiding errors.

- Data Analysis. The EOSE will process and present spacecraft data in a manner which will permit test personnel to easily, quickly, and accurately analyze the data and make judgments accordingly. The STC, through its computer facility, has the capability to process and reduce spacecraft data and present it to the test personnel in a format which is readily comprehensible. Data will be displayed and printed in engineering units.

- Safety. The EOSE will incorporate design and testing features providing safe operating conditions for test personnel, the spacecraft, and the EOSE. Specific examples are: 1) Special markings on dangerous or critical equipment areas; 2) cabling connectors keyed to prevent improper matings; and 3) verified test procedures for all test and trouble-shooting operations.

- Self-Test. The EOSE will provide, wherever practical, self-testing circuitry for the hardware and self-test diagnostic routines for the computer.

11.2 DESIGN CRITERIA

Basically, the criteria for the design of the EOSE is that it support the spacecraft during test and integration, and that it supply stimuli to and process data from the spacecraft. In more specific terms, the criteria are:

- Power Support. The EOSE will supply raw bus power to the spacecraft simulating the output of the solar arrays.

- Data Recording. All telemetry data, command transmissions, and the essential EOSE status data will be recorded on magnetic tape. This recorded data will be time-correlated in that the facility time code generator output will also be recorded. Command transmissions (time-related) will additionally be printed on an off-line printer. The data recording will be accomplished on the two analog recorders supplied. Data history records will be recorded on magnetic tape providing trend data on the performance of each subsystem aboard the spacecraft throughout the entire test cycle. The recording functions here are provided by the STC computer.

- Transportability. The EOSE must withstand environments encountered while being transported from one location to another by common carrier. For the most part, the equipment will be housed in standard single-bay relay racks. Some two- and three-bay racks will be used, but nothing larger than three bays will be required.

- EOSE Status Monitoring. Discrete voltage levels will be provided to the computer from relay contacts in all the stimulus equipment. These levels will indicate the status of all switchable functions in the stimulus equipment.

- Automatic Stimulus Control. All switchable functions in the STC which control stimuli (inputs) to the spacecraft will be controllable by the computer. The stimulus equipment will have control over manual/automatic mode selection. If the stimulus equipment is switched to the manual mode, the computer control will be overriden to preclude automatic control. However, if the stimulus equipment is switched to the automatic mode, test personnel will be able to switch to the manual mode.

- Automatic Sequencing. The EOSE will provide automatically controlled spacecraft operation through preprogrammed test sequences (subject to manual override). Preprogrammed command sequences are executed by the computer through the command encoding and transmitting equipment. Responses are checked by processing the data returned from the spacecraft through the EOSE receivers and decommutation equipment. Hard-line data from the spacecraft is also processed. The STC stimulus equipment can also be controlled automatically by the computer to provide additional spacecraft control not available through commands. Controls for manual override are included in the test conductors console.

- Manual Control. The EOSE will provide for manual control as a backup to the automatic mode and for special tests. All the inputs required to operate the spacecraft can be supplied manually. This capability permits spacecraft operation if the computer malfunctions or when the test personnel wish to examine closely the operation of selected spacecraft systems.

- Telemetry Processing. The STC will process spacecraft engineering data including status information and science data simultaneously. Two separate channels of demodulation, bit synchronization, and decommutation equipment are included in the STC to permit entry of each type of data into separate memory banks of the STC computer. The memory banks operate independently, permitting simultaneous and asynchronous access.

- Safe Operation. The EOSE will provide safeguards against erroneous activation of critical circuits (explosive devices, deployment, etc.). Both hardware and software safeguards will be included to prevent generation of commands or other stimuli which could result in damage to equipment or injury to personnel.

- Electromagnetic Compatibility. The EOSE will incorporate any bonding and grounding practices necessary to assure electromagnetic compatibility both within the EOSE and with the spacecraft. Critical functions will be housed in racks which are shielded against radiation of electromagnetic energy.

- Decontamination. The racks of the EOSE which are to be physically located in the spacecraft area will be mechanically designed such that they may be easily cleaned with a vacuum cleaner. Further, these racks will not exhaust cooling air to the spacecraft room except through a suitable filter. The stimulus equipment for the guidance and control subset will be designed such that they can be easily cleaned with a vacuum cleaner. There will be no requirement for EOSE to withstand ETO environment.

12. EOSE SYSTEM FUNCTIONAL DESCRIPTION

12.1 SYSTEMS TEST COMPLEX UTILIZATION

First and foremost the STC is designed to support system testing of the Voyager spacecraft. This design discipline can be extended with relatively minor modifications to include subsystem testing. The basic use, however, will never be compromised to accommodate a lower level of testing.

The spacecraft equipment will arrive at the system integration facility fully tested and certified by factory test equipment completely separate from the STC. These units are assembled into panels which are to a large degree functional in nature. At this point the STC is first used in support of the integration and test of these panels. Table 12-1 lists the panels which go to make up the spacecraft and the various units which make up the panels. After the units have been assembled onto panels and checked out, the panels will be integrated onto the spacecraft. Here again the STC will play the major role of support during the integration phase. The STC will supply the interfaces necessary to operate the spacecraft, admittedly in a degraded mode, without all its panels installed. When all the panels have been integrated and the STC assumes its intended role of system test equipment, the spacecraft is complete.

System testing will begin with a series of tests on the engineering model spacecraft. The series of system tests performed on the engineering model spacecraft are particularly important in regard to the STC, since these tests will serve to validate the design of the STC. These tests will also develop experience and confidence in the use of the STC. In addition to its primary use as a test instrument for the spacecraft, the STC has other auxiliary benefits. For instance, since it incorporates some equipment which is also incorporated in mission-dependent equipment (MDE), the STC serves as a test bed to verify the design of these

Table 12-1. Spacecraft Panels and Panel Units

Equipment Panel I
- Batteries (3)
- Limb Crossing
- Terminator Crossing Detector

Equipment Panel IV
- S-Band Receivers (4)
- Preamplifiers (4)
- Diplexers (4)
- Circulator Switch Assembly
- Baseband Assembly
- DC-DC Converter (2)
- Relay Link Receivers (2)
- Relay Link Demodulators (2)
- Transmitter Selector
- Receiver Selector
- Low-Gain Antenna Selector
- Modulator Exciters (2)
- One-Watt Transmitter
- Power Amplifier and Power Supply (2)
- Junction Box
- Limb Crossing Detector
- Terminator Crossing Detector
- Antenna Drive Electronics

Guidance and Control Panel A
- Inertial Reference Assembly
- Canopus Sensor
- Fine Sun Sensor
- J-Box

Equipment Panel III
- Spacecraft and Capsule TV Recorders (4)
- Engineering and Science Recorders (2)
- Telemetry Data Processor
- DC-DC Converter (2)
- Junction Box

Equipment Panel VII
- Primary Computer and Sequencer
- Backup Computer and Sequencer
- Command Decoder (2)
- Bit Synchronizers (2)
- Shunt Assemblies (2)
- DC-DC Converters (2)
- Junction Box
- Guidance and Control Electronics Assembly

Equipment Panel VIII
- Power Control Unit
- Distribution Control Unit
- 400-Hz Inverter
- DC-DC Converters (4)
- Pyrotechnic Control Unit
- Junction Box
- Shunt Assembly (2)

Guidance and Control Panel B
- Canopus Sensor
- Fine Sun Sensor
- J-Box

Equipment Panel V
- Junction Box, Science
- DC-DC Converter (2)

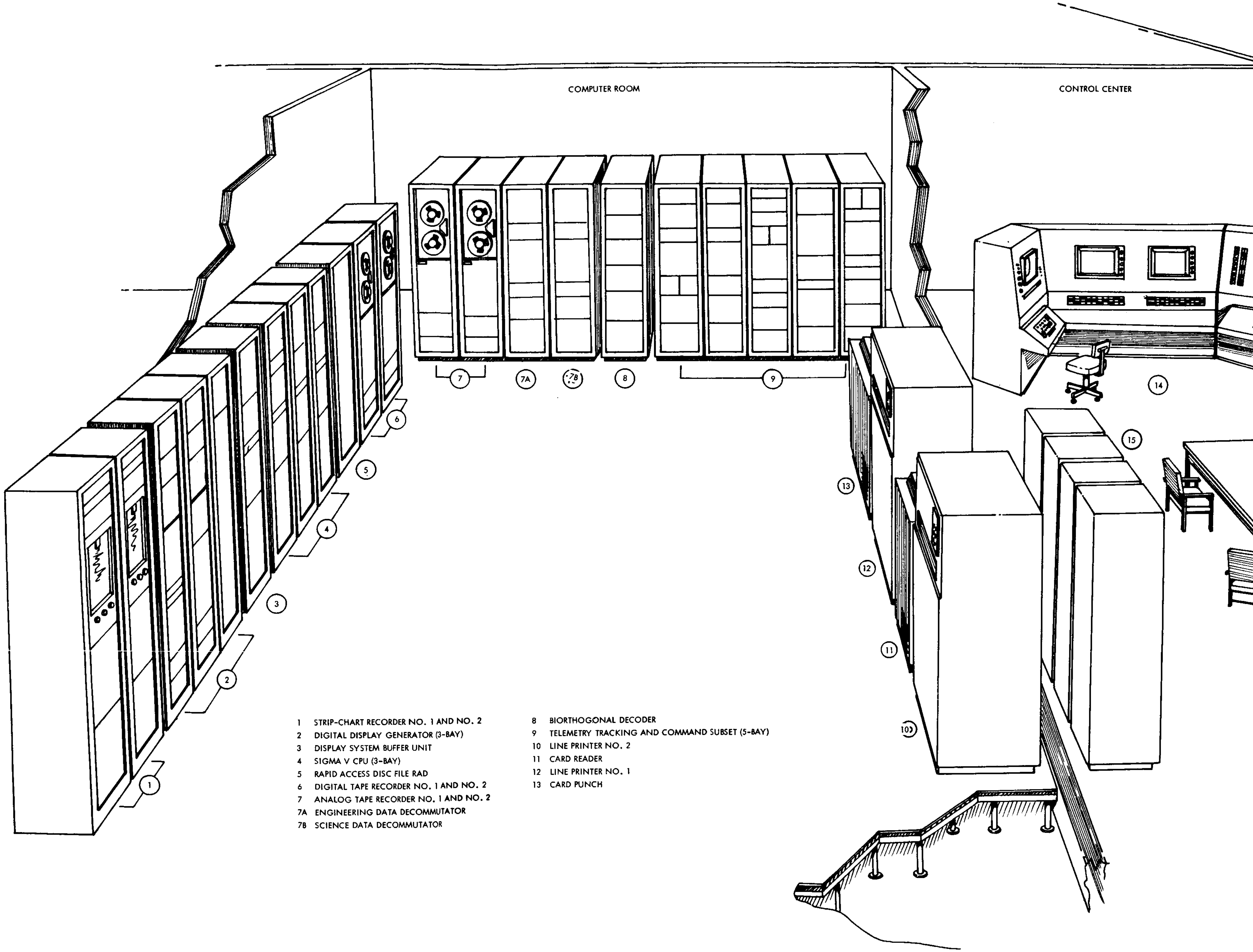
items of equipment before their use is required in the MDE. Since the computers in the DSN stations and the STC computers are expected to be compatible, some of the MDE software can be checked out using the STC. Figure 12-1 shows a pictorial layout of an STC as intended for use at TRW during integration and system testing.

A total of five STC's will be developed, for use at TRW, launch facility, and compatibility testing at the DSN station at Goldstone, California. Each of these STC's will be identical with very minor differences to account for different conditions at the different test sites. For instance, the spacecraft will be installed in its shroud at the launch facility and a special interface unit to mate with the shroud umbilical will have to be developed. Another difference will be the possibility of installing small stimulators inside the shroud to supply stimulus to the sensors after the spacecraft is "buttoned up" and ETO has been pumped in. These items and any special items needed to support test at the DSN station are, however, minor in relation to the STC in total.

12.2 SYSTEMS TEST COMPLEX HARDWARE

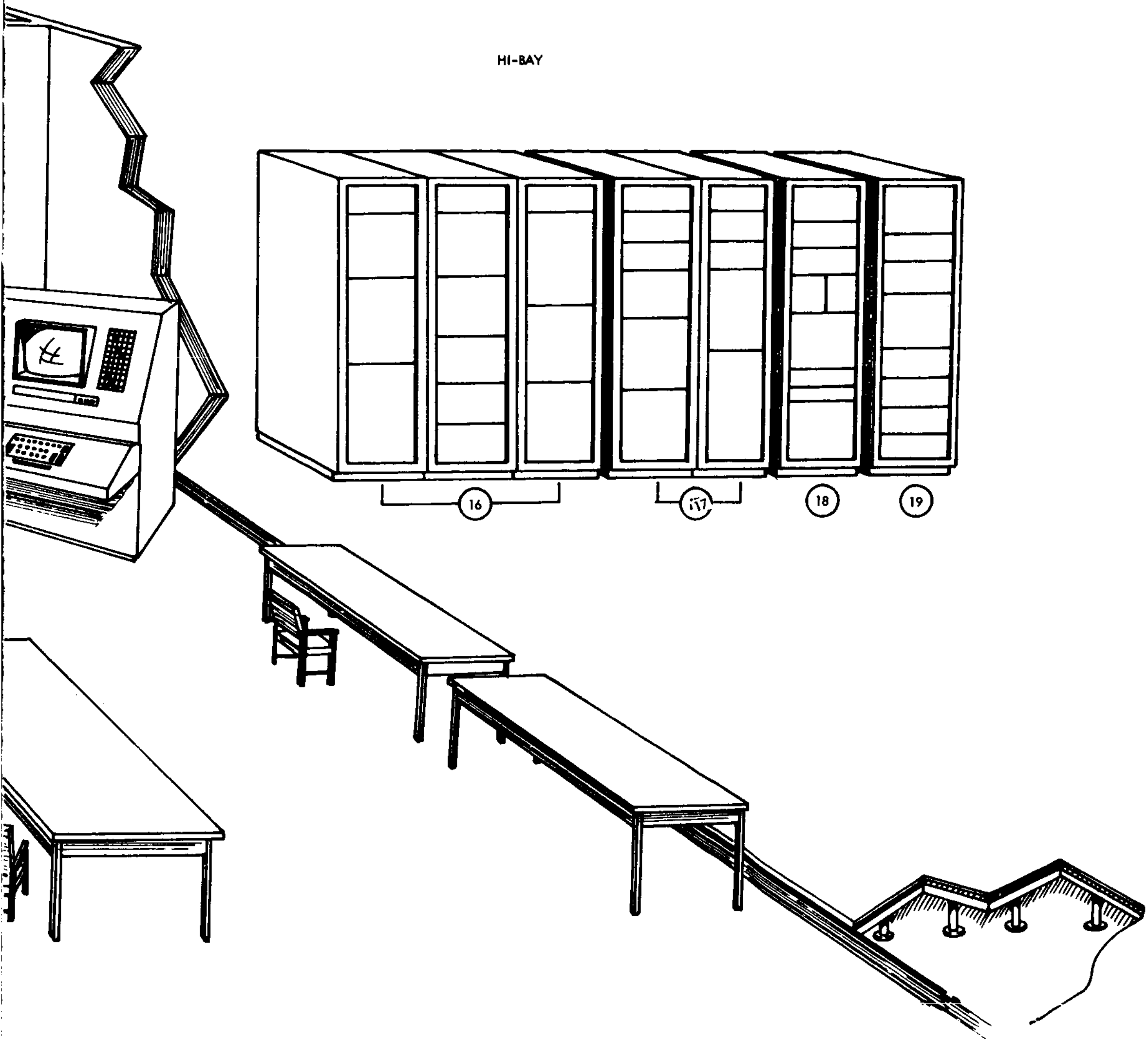
All the electrical equipment required for system level testing of the spacecraft will be integrated into a complex of equipment referred to as the system test complex (STC). Through a combination of hardline and electromagnetic links with the spacecraft, the STC is able to stimulate the various subsystems of the spacecraft and monitor spacecraft responses. The STC is divided into the following functional equipment groupings:

- Hardline/peripheral subset
- Telemetry, tracking, and command subset
- Data processing subset
- Guidance and control subset
- Science equipment subset
- Interface simulation subset.



FOLDOUT FRAME 1

FOLDOUT FRAME 2



14 TEST CONDUCTORS CONSOLE
15 TV RECONSTRUCTION EQUIPMENT
16 HARDLINE/PERIPHERAL SUBSET (3-BAY RACK)
17 STABILIZATION AND CONTROL SUBSET (2-BAY RACK)
18 SCIENCE EQUIPMENT SUBSET
19 INTERFACE SIMULATION SUBSET

FOLDOUT FRAME 3

Figure 12-1
SYSTEMS TEST COMPLEX in conceptual layout for system integration and test activities at TRW, has three main areas; the spacecraft is located in the Hi-bay, hygienically isolated from the rest of the equipment. Similar complexes will be set up at KSC, Goldstone, and White Sands.

A functional block diagram showing the relationships among these subsets and their interfaces with the spacecraft is shown in Figure 12-2. Each of the subsets is defined more completely in subsequent paragraphs.

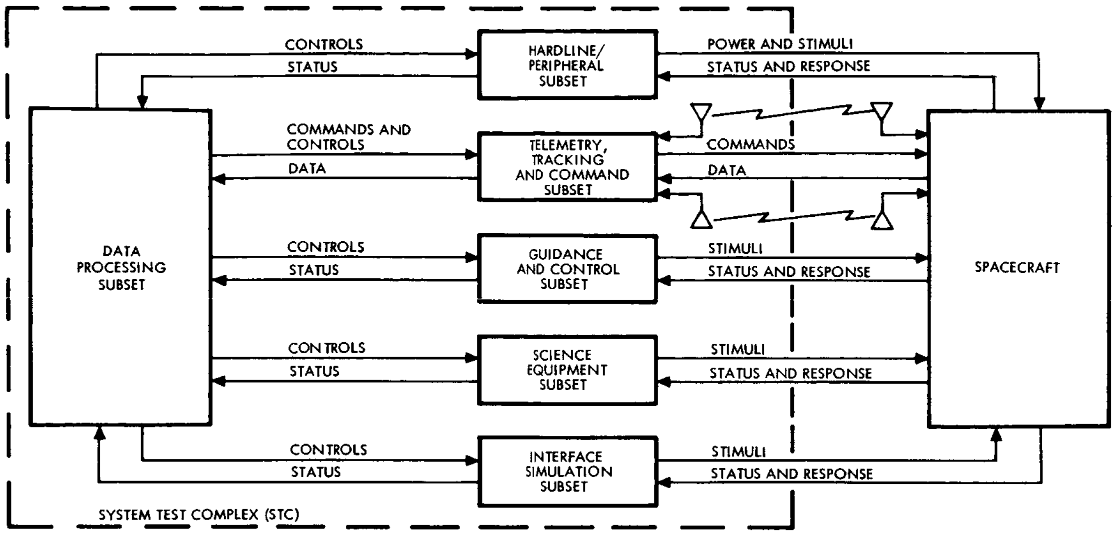


Figure 12-2
EOSE SYSTEM TEST COMPLEX (STC) shows relationships among the subsets and their interfaces with the spacecraft, indicating control and status signal flow with the data processing subset and the flow of stimulii to the spacecraft and its response and status signals.

12.2.1 Hardline/Peripheral Subset Functional Descriptions

The hardline/peripheral subset provides ground power to the spacecraft, incorporates capability to control and monitor the spacecraft power subsystem, has the capability to test spacecraft ordnance circuits, and includes controls and monitors for other miscellaneous spacecraft parameters. The hardline/peripheral subset also has an interface with the computer so that the outputs may be put under computer control and the monitor points may be entered into the computer.

Figure 12-3 shows the conceptual rack layout of the hardline/peripheral subset. The functions of the individual units are as follows:

- Frequency Counter: provides a means for monitoring the 400-Hz inverter output of the power subsystem and the sync frequency, also generated by the power subsystem.

- Power Distribution Status Display: displays various status indications relating to the operation of the spacecraft power subsystem.

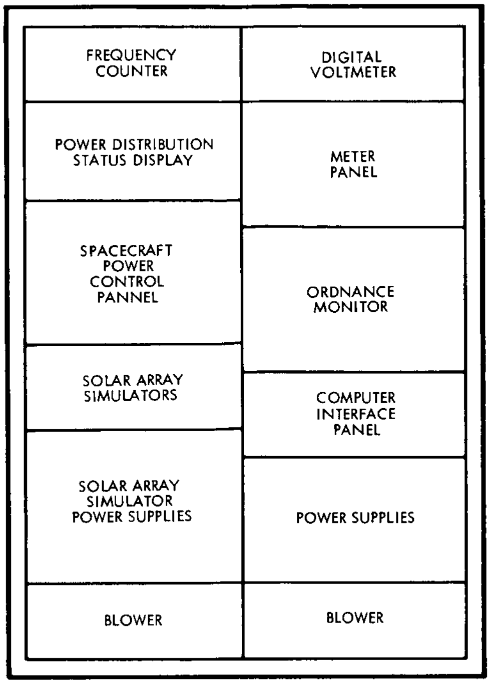


Figure 12-3

RACK LAYOUT OF THE HARD LINE/PERIPHERAL SUBSET depicts the physical relationship of the various drawers comprising the subset.

- Spacecraft Power Control: provides manual control for all ground power operation. Provisions are included to permit computer control in an automatic mode.

- Digital Voltmeter: provides continuous monitoring of any one of a number of selectable parameters. It can monitor both DC and AC functions.

- Meter Panel: contains four voltmeters, four ammeters, and a temperature meter. The temperature meter can be switched to observe the temperature of any one of the three spacecraft batteries. One of the voltmeters measures bus voltage while the other three measure the voltage of each of the batteries. Each voltmeter has an associated ammeter.

- Computer Interface Panel: provides status indications to the computer and accepts commands from the computer. In the automatic test mode this panel supplies the communications necessary for the computer to operate the hardline/peripheral subset. In the manual mode this panel is disabled.

- Solar Array Simulator: accepts power from the solar array simulator power supplies, conditions this power to precisely resemble the output of a solar array, and supplies this power to the spacecraft. The output of the solar array simulator will be variable such that all expected operating characteristics of the solar array can be simulated. This involves generation of I/V characteristics varying from 1 AU at 107°C to 1.67 AU at 23°C for the insulated arrays and from 1 AU at 75°C to 1.67 AU at 2°C for the uninsulated arrays. Figure 12-4 shows the simplified block diagram of the solar array simulator. The open circuit voltage and short circuit current determine the terminal points of the I/V characteristic. The series resistor determines the slope of the voltage portion of the curve. The knee of the curve and the slope of the current portion are controlled by the R-C compensation networks on the differential amplifiers. There will be three of these solar array simulators for the input to the linear shunt element and two simulators for each of the saturated shunt elements.

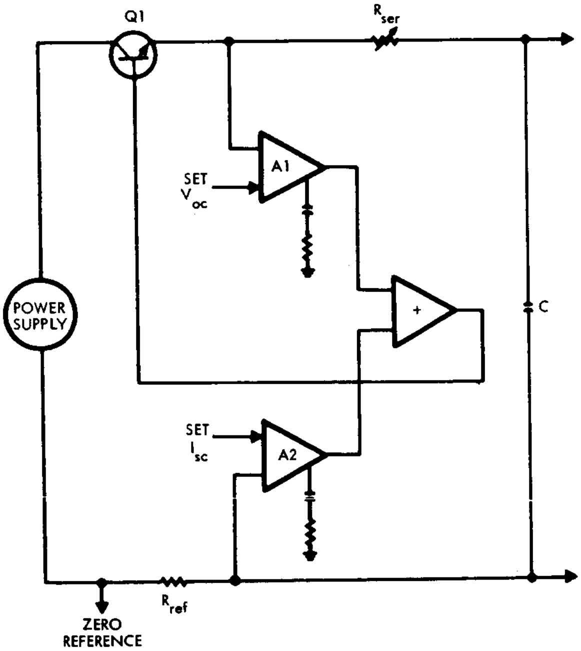


Figure 12-4

EOSE SOLAR ARRAY SIMULATOR has amplifiers A1 and A2 driving the error signals which drive the regulating thermistor Q1. R_{ref} generates the current error signal and R_{ser} sets the output serial resistance.

- <u>Ordnance Monitor:</u> supplies a simulated ordnance device to the firing circuit, tests the output of the firing circuit for proper timing and firing levels, and monitors the firing circuits for transients of sufficient energy to possibly fire an ordnance device. Figure 12-5 shows the block diagram of the ordnance monitor circuit. An amplifier labeled Signal Present starts the testing sequence whenever a signal with an energy level above the "no fire" level for an ordnance device appears across the simulated

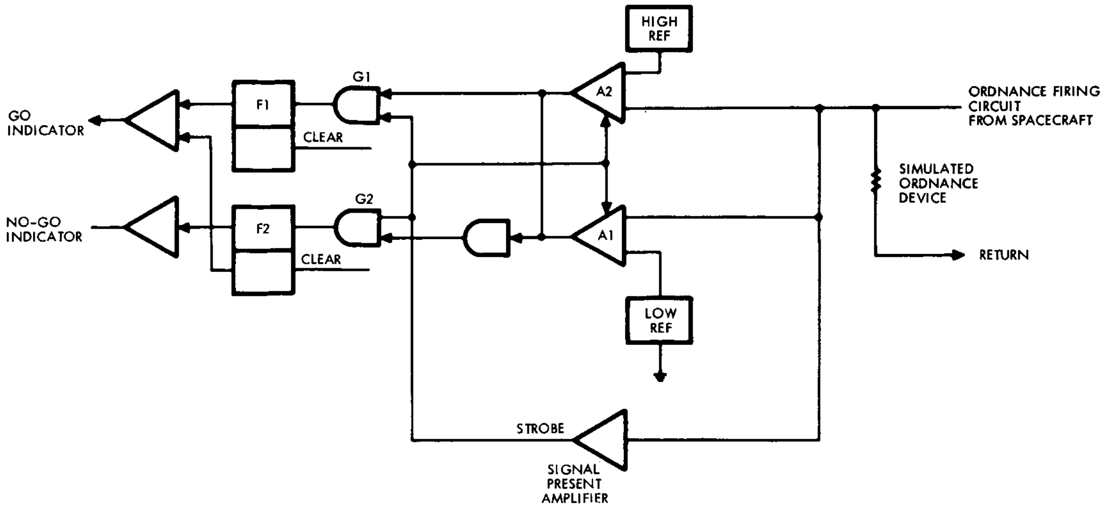


Figure 12-5
ORDNANCE MONITOR CIRCUIT supplies a simulated ordnance device, tests the firing signal for proper level in amplifiers A1 and A2, tests for proper timing through gates G1 and G2 and holds the results in flipflops F1 and F2.

device. The "fire" signal is tested for its "all fire" capability by amplifier A1. Amplifier A2 assures that a signal of too-high amplitude, as would be caused by an open return line, is also flagged as No Go. The output of the amplifiers is directed through timing logic to a pair of flip flops which light either a Go or a No Go indicator. With two separate indicators a light is always on regardless of the results of the test and one state is not inferred from the lack of indication of the other. The indicators are cleared either manually or by the computer when under automatic control.

12.2.2 Telemetry, Tracking, and Command Subset Functional Description

The telemetry, tracking, and command subset is used to test, evaluate, and monitor the performance of the spacecraft uplink and downlink equipment. It includes standard commercial equipment plus special test equipment designed specifically for the Voyager spacecraft. A similified block diagram of the TTCS is shown in Figure 12-6. The TTCS comminicates with the spacecraft through both hardline and radiated links.

The various parameters which will be measured by the TTCS are listed in Table 12-2. In addition to these measurements the TTCS generates commands to the spacecraft either under manual or computer control. It also monitors all transmitted commands, decodes the

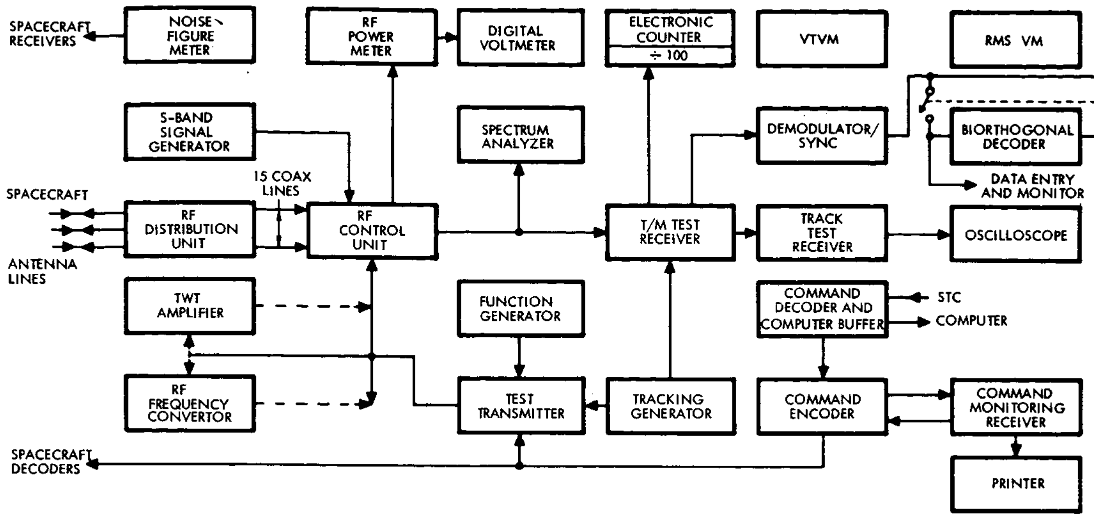


Figure 12-6
TELEMETRY, TRACKING, AND COMMAND SUBSET shows the simplified signal flow and functional interfaces between the various units of the TTCS.

Table 12-2. Tests Which will be Performed by the Telemetry, Tracking, and Control Subset

A. Tests on Spacecraft Transmission Parameters

1) Power output
2) Output frequency
3) Modulation sensitivity
4) Modulation index
5) Incidental FM and phase jitter
6) Spurious outputs

B. Tests on Spacecraft Receiving Parameters

1) Input noise figure
2) Frequency acquisition bandwidth
3) Phase-lock loop acquisition level
4) Phase-lock loop bandwidth
5) Out-of-band signal rejection
6) Demodulation linearity

C. Other Tests

1) Command decoder threshold
2) Zero range tracking delay
3) Receive-transmit frequency coherence
4) Telemetry and tracking crosstalk
5) Biorthogonal decoding at threshold
6) STC self-tests

commands and inputs the monitored command to the computer. Commands can be sent to the spacecraft by either hardline or radiated link. The command monitoring function is performed in either case. The TTCS also includes demodulator/synchronizer units to perform demodulation and bit synchronization on telemetry data prior to sending it to the telemetry decommutation equipment in the DPS.

In addition, a biorthogonal decoder is provided for use in testing and spacecraft encoder and downlink communications near threshold. In this test mode the coded data is fed directly from the demodulator/synchronizer channel to the biorthogonal decoder. The biorthogonal decoder is designed primarily for use within the Deep Space Net. A complete description of the equipment can be found in Section 6. 4.

A conceptual rack layout of the TTCS is shown in Figure 12-7. Each of the items is discussed below.

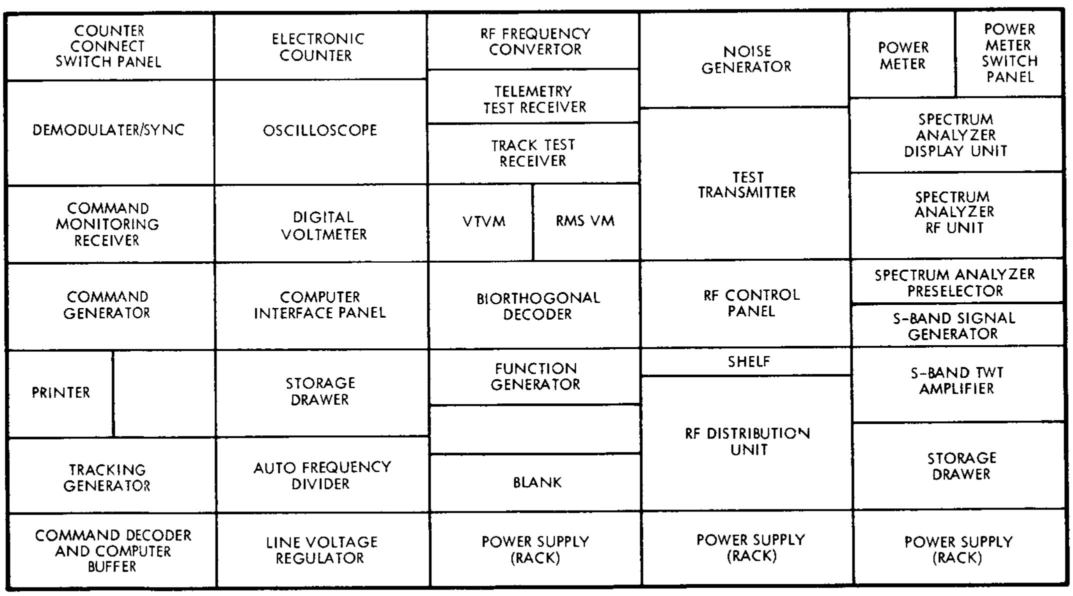


Figure 12-7
RACK LAYOUT OF THE TT AND C subset shows the proposed physical layout, except that this subset will be built in two or three separate racks, rather than one unit.

12. 2. 2. 1 Electronic Counter

The automatic frequency divider, the electronic counter with self-ranging plug-in, and the associated switch panel comprise a completely automatic frequency counting system with measurement capability from DC to 12.4 gHz. No tuning or adjustment is required to perform these measurements. Coax switches which can be controlled manually or by the computer select the desired input.

12.2.2.2 Demodulator/Synchronizer

The demodulator/synchronizer for spacecraft telemetry Link 1, 2, and 3 are identical to the MDE assemblies provided to accomplish the functions of data demodulation and synchronization at the DSIF stations. These units operate with the EOSE in the same manner as their operational counterparts by providing reconstructed noise free data from the PCM bit streams in the case of uncoded data. Likewise when the data is biorthogonally encoded the demodulator/synchronizers interface with the biorthogonal decoder to perform the complete operation of data recovery and decoding in the same manner as for the MDE. A full description of these assemblies is contained in Sections 6-1 through 6-4.

12.2.2.3 Command Encoder

The primary function of the command encoder is to generate Planetary/Vehicle command signals of three types in a form suitable for modulating the command transmitter. Normal operation of the command encoder is under the control of the computer. As a backup capability an emergency operating mode exists where in the command encoder will provide a manual capability for the initiation of all three types of Planetary/ Vehicle command signals. The command encoder is identical to the MDE assembly which is fully described in Section 6.6.

12.2.2.4 Printer

This is a low-speed character printer which provides a printed record of every command emanating from the command encoder.

12.2.2.5 Command Decoder and Computer Buffer

This unit decodes commands received through the command monitor receiver or from a hardline to the command encoder. The commands are decoded and buffered for entry into the computer in the command verification process.

12.2.2.6 Command Monitor Receiver

This unit monitors commands transmitted to the spacecraft.

12.2.2.7 Tracking Generator

This unit produces the pseudo-noise-coded, binary ranging words to be transmitted to the spacecraft for zero-range delay and crosstalk tests. This data is also supplied to the tracking test receiver for range measurements.

12.2.2.8 Digital Voltmeter

The DVM will accept both AC and DC inputs and has an output which can be entered into the computer for monitoring purposes.

12.2.2.9 RF Frequency Converter

This unit is used in STC validation and self-tests. It has the capability to translate the output of the test transmitter from the uplink frequency to the downlink frequency. This translated signal can then be fed into the telemetry test receiver. The translation is made in the ratio of 240/221. In this manner, computer-generated patterns can be sent through the test transmitter and brought back into the computer by the telemetry decommutation equipment for comparison. Command generation and verification can be tested in a similar manner.

12.2.2.10 Telemetry Test Receiver

In addition to accepting and demodulating the spacecraft RF signal, this receiver provides measurements of incidental FM, phase jitter, and downlink tracking crosstalk on the received signal.

12.2.2.11 Tracking Test Receiver

This receiver compares two PN coded data streams, one from the tracking generator and the other from the spacecraft. Phase interval between identical ranging words in the two streams is measured and displayed for ranging purposes.

12.2.2.12 VTVM and RMS VM

These are general-purpose instruments.

12.2.2.13 Function Generator

This unit supplies the test transmitter with various waveforms such as those used for loop bandwidth, doppler rate, and demodulation tests.

12.2.2.14 Noise Generator

This unit is used in tests of the spacecraft receivers. These tests are performed periodically to establish trends and margins.

12.2.2.15 Test Transmitter

This unit is used to transmit commands to the spacecraft; however, is can be used for several kinds of tests on spacecraft equipment. Such tests include frequency acquisition bandwidth, loop acquisition level, and uplink/downlink frequency coherence. With proper modulation it can also be used for tests of loop bandwidth, demodulation linearity, and command decoder threshold.

12.2.2.16 RF Control Panel and RF Distribution Unit

These units contain the controls for and the directional couplers which interface via hardlines with the spacecraft TWT's. All couplers are terminated to prevent the TWT's from operating into no load as a result of switch failure or human error in switch positioning. The couplers provide power reduction and isolation for signals from the TWT's.

12.2.2.17 Power Meter

This unit is used to measure power from the spacecraft transmitters as well as test transmitter output power.

12.2.2.18 Spectrum Analyzer

This unit is used in tests of modulation index, spurious outputs, and crosstalk.

12.2.2.19 Signal Generator

This unit is used to test spacecraft receiver rejection levels and to determine losses within the tracking, telemetry, and command subset RF distribution system.

12.2.2.20 TWT Amplifier

This unit is used for amplifying transmitted power from the tracking, telemetry, and command subset.

12.2.3 Data Processing Subset Functional Description

The data processing subset includes the telemetry decommutation equipment, the computer complex, the test conductors console, analog tape recorders, strip chart recorders, status displays, and video reconstruction equipment. The data processing subset has the capability for decommutating and processing telemetry data, transmitting commands and stimuli to the spacecraft (through other subsets), recording telemetry and test data, and displaying spacecraft status information. This equipment, in addition to spacecraft testing and launch operations, will be used for preparing spacecraft test and analysis programs, the spacecraft engineering data calibration information, and the listing of spacecraft test procedures. The last two items will be done in an off-line mode utilizing a media-to-media data transfer routine.

Data Processing Subset Equipment Descriptions

Figure 12-8 shows a functional block diagram of the data processing subset. The equipment is described in more detail below.

Telemetry Decommutation Equipment. This equipment receives serial data input from either the demodulator in the telemetry tracking, and command subset; hardline directly from the spacecraft under test; or playback from a previously recorded analog tape or from the programmable telemetry format simulator, which is an auxiliary component of the decommutation equipment.

The decommutation equipment will perform synchronization (bit, word, and frame), data formating, serial-to-parallel conversion, buffering of the input data for transmission to the computer, digital-to-analog conversion of up to 16 selectable values for output to strip chart recorders, and, upon manual selection, data compression.

The purpose of data compression is to reduce the amount of data to be processed by the computer through the elimination of data that provides no new information.

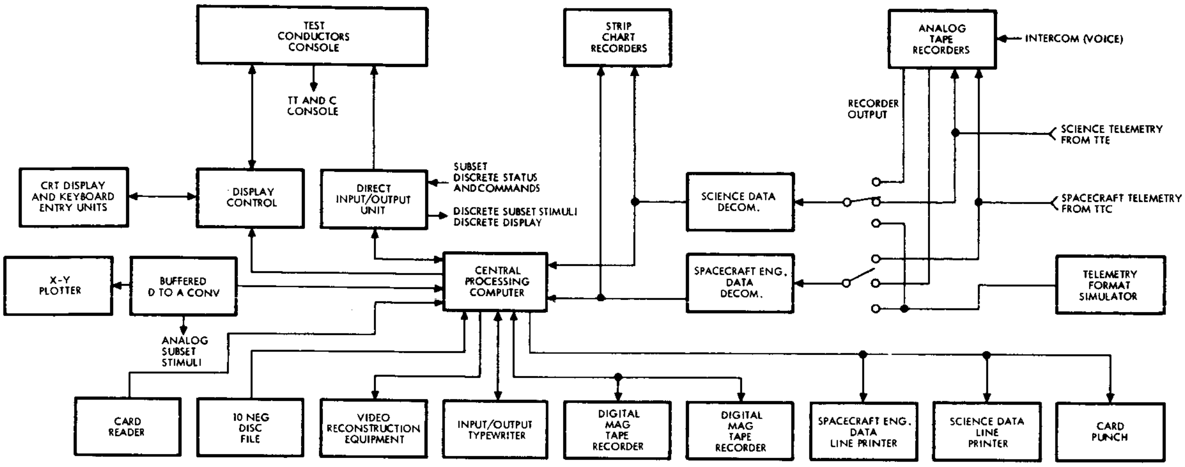


Figure 12-8
DATA PROCESSING shows data flow from the two telemetry lines through decommutation equipment into data processing equipment and out to the various display devices including printers, CRT displays, X-Y plotters, and discrete displays.

Data compression is performed by comparing the present value of a telemetry word with the previous value of the same word. If the values differ, the word is input to the computer and the present value replaces the previous value; otherwise it is ignored. This technique requires storage for all previous values of telemetry words.

Since there are two different but simultaneous telemetry transmissions from the spacecraft, two sets of decommutation equipment are required. One is necessary for the scientific data, the other for the spacecraft engineering data. Each set of decommutation equipment shall be directly interchangeable.

The programmable telemetry format simulator, only one of which is required, is also part of the decommutation equipment. This unit provides for closed-loop checkout and maintenance of the decommutation equipment, a source of calibration signals for the strip chart recorders and a controlled data source for development of spacecraft test and analysis programs.

Video Reconstruction Equipment. The basic hardware used with the MDE for image reconstruction is also integrated into the STC to provide a means of testing the spacecraft imaging subsystem during systems integration and test and displaying test patterns on a visual display scope.

None of the equipment or software associated with scan conversion or picture enhancement will be provided for the STC. The capability will exist to photograph the visual scope displays for off-line analysis and, in addition, magnetic tape recordings will be made of the video data during system tests involving the imaging subsystem. If the necessity arises these tapes may be taken to the SFOF for complete processing of data including scan conversion and picture enhancement to ensure complete compatibility with the operational ground processing laboratory at an early point in the program.

Complete processing of the engineering data associated with the spacecraft imaging subsystem will be provided within the STC. The computer will strip the engineering data from the remainder of the video data stream and convert the data to a useful format for test purposes.

A full description of the ground hardware used in the image reconstruction process is included in Section 6.5.

Analog Magnetic Tape Recorder. The analog magnetic tape recorder, two of which are required, will be capable of recording seven channels of information and a voice channel at speeds of 1-7/8, 3-3/4, 7-1/2, 15, 30, and 60 inches per second. The speed selection will be used to provide a playback time scale expansion or reduction capability to ensure presentation of the data on an optimum time base relative to the display device used (i.e., strip chart, X-Y plotter, etc.).

The information to be recorded is two channels (one redundant) of spacecraft engineering telemetry data, two channels (one redundant) of scientific telemetry data, one channel of spacecraft command transmission, one channel of the TRW facility time code, and one channel of servo control information to provide for transport capstan drive speed accuracy at all record/playback speeds. The voice channel will be recorded on the tape edge and will provide a running commentary of all test operations. The inputs to the voice channel will be from all test stations via the system intercom network.

Computer Complex. This equipment consists of a Scientific Data System Sigma 5 general purpose computer with 32, 768 32-bit words of memory, input-output typewriter, two digital magnetic tape recorders, two line printers, one card reader, one card punch, one 10-megabit disc file, a digital-to-analog converter, and an X-Y incremental plotter.

- Central Processing Computer. This unit will control processing of telemetry data, generation of spacecraft command sequences, output of display and printer information, monitoring of the status of other subsets through discrete inputs, and generation of discrete outputs to control other subsets.

- Input-Output Typewriter. This unit provides the man-machine interface which enables the computer operator to communicate with the central processing computer. It is utilized to provide mode and control function inputs to the central processing computer..

- Line Printers. Two buffered line printers are required, one for printing scientific data and one for printing spacecraft engineering data. The line printer output will provide permanent documentation of test data, formaced to enable rapid analysis. All copies of printer output will be annotated as to date, time, type of test, spacecraft serial number, etc. The line printers will be used during computer program preparation to provide program listings, error messages, and memory content listings. The line printers operating in an off-line mode will be used to provide spacecraft engineering calibration tabulations and test procedure listings.

- Card Reader. The card reader is utilized as the primary media for inputting vendor-supplied or TRW-developed computer programs. It is utilized to input source decks during computer program preparation and checkout operations. During test operations, the card reader provides a means of inputting test sequence parameters, control functions, and spacecraft calibration criteria into the computer memory. The card reader is used to input calibration data and test procedure information into memory from which a hard-copy printout can be made using the digital line printer.

- Card Punch. The card punch is used in an off-line mode to prepare punched card decks utilizing the central processing computer or any of its associated peripheral equipment as the information source. The card punch will significantly reduce the time required to prepare source decks of test sequence parameters, control functions, calibration criteria, etc.

- Digital Magnetic Tape Units. These units and their associated controller are used to record pertinent spacecraft engineering and scientific test data, subset status data, and spacecraft command information during spacecraft test operations. Tape recordings are also utilized for storage to enable efficient and rapid off-line operations on previously recorded test data. During program preparation and assembly, the assembler program is kept on one tape unit while the program being assembled is written on the other tape unit.

- Disc File. The disc file is the bulk memory unit of the computer system. All computer library programs and routines (i.e., real-time monitor, system diagnostics, compilers, assemblers, driver routines, math routines, etc.) are stored on the disc file.

- Display Control. This unit is composed of certain basic elements of the Scientific Data Systems Sigma 2 general purpose computer. The display control is the external buffering device for the six cathode-ray-tube displays. The display control unit processes all requests for information, formats the data from the central processing computer, and outputs it to the cathode-ray-tube displays. The display control unit will have the memory capacity necessary to store all spacecraft and subset information which could be requested by an operator through any of the six remote displays. In addition to the two cathode-ray-tube displays in the test conductors console, there will be one each in the peripheral hardline unit; the data analyst's operating position; tracking, telemetry, and command subset; and one available for use by the spacecraft experimenters.

- Digital-to-Analog Converter. The digital-to analog converter will be used to provide analog stimuli for closed-loop test functions of the spacecraft. It will also provide the input to the incremental X/Y plotter. The digital-to-analog converter will contain its own buffer to ensure optimum computer usage.

- Incremental X-Y Plotter. The X-Y plotter, during spacecraft testing, will be used to provide permanent records of rapidly changing telemetry measurements. In an off-line mode the X-Y plotter will be used to provide graphs of spacecraft engineering calibration data. It will also be used to provide a permanent record of spacecraft subsystem performance. This will be one of the outputs from the data logging program and will be used to analyze and diagnose performance trends of spacecraft subsystems.

- Direct Input-Output Unit. This unit will provide the necessary interface and buffering to output discrete spacecraft and subsystems status information and discrete stimuli to the spacecraft and other subsets. Spacecraft commands and subset status information will be input to the computer via this unit.

Test Conductor Console. This unit provides a centralized location from which tests on the spacecraft will be controlled and monitored. It includes two cathode-ray-tube display and keyboard entry units, subset equipment status display, spacecraft status display, manual spacecraft command selection and generation controls, and alarm and warning indicators. The alarm indicators result from the detection of abnormal spacecraft or subset equipment conditions which would lead to equipment damage or degradation if the test continued. The detection of an alarm condition, in addition to displaying the information, would automatically issue the spacecraft commands or subset equipment stimuli to power down or disconnect to reduce the possibility of equipment damage. A warning indication would only alert the test conductor to a condition which, although not hazardous, should be realized prior to proceeding with the test. All alarm and warning signals will be under computer-program control, and the detection threshold parameters which will be unique for each test will be input via the system card reader.

CRT Display System. Two different types of display systems have been investigated. The first system considered was the standard cathode-ray-tube display and keyboard entry equipment supplied by the computer manufacturer with all required equipment for computer interface. The system can display up to 32 lines of alphanumeric characters (32 characters/line). With this system and proper programs, fixed formats could be displayed or several parameters could be chosen for display be test personnel. The test personnel could change the selection through the keyboard entry unit, could inhibit the display of any parameter, and, in general, could exercise complete control of the data displays within the programming and character capacity limits.

Discussions with MSFC personnel indicated a desire for additional display capability, particularly the capability to display analog waveforms. Subsequent investigation led us to a system developed by Hazeltine and currently in use in the Central Instrumentation Facility of KSC. This system has all the basic capability of the standard computer display system discussed above and has additional capability for displaying analog waveforms, vectors, and characters of varying size. A general description of this system follows.

- The purpose of the digital display generator (DDG) is to accept digital data from an external buffer (the Sigma 2) and to convert this data to television video signals which, when applied to standard 525-line television monitors, produce alphanumeric displays. The DDG produces alphanumerics, vectors, lines, and background grids, the latter at half the brightness of the other data. Ten video channels, each carrying different display data, are supplied to the monitors.

  Figure 12-9 is a drawing of the digital display generator. It consists of three cabinets and a magnetic drum. The electronics are contained in six vertical slide-out drawers. Five of the drawers contain eight rows of printed circuit boards each; the sixth drawer contains a core memory and associated boards. All boards utilize transistor circuitry. A meter panel and standard commercial power supplies occupy the upper portion of each cabinet. A power control panel occupies the lower portion of each cabinet. The three cabinets are called the core memory cabinet, the data processor cabinet, and the video processor cabinet.

- A system block diagram is shown in Figure 12-10. A buffer furnishes digital words defining the information to be displayed to the digital display generator. The DDG accepts these words in the data processor, where the words are converted into television picture elements and assembled into a complete television picture in the core memory.

Figure 12-9
HAZELTINE DIGITAL DISPLAY GENERATOR (DDG) consists of three racks containing magnetic drum memory, channel logic, and time shared common logic. The CRT displays are not shown here. This unit has the capability of driving 10 CTR's.

- This picture is composed of 1024 video elements for each of 480 active TV lines. When completely assembled, the picture is transferred to a designated section of the drum memory, which is used as the television refresh storage. Pictures for 10 different television displays are formed in sequence in the core memory and each is then stored in a separate section of the drum memory.

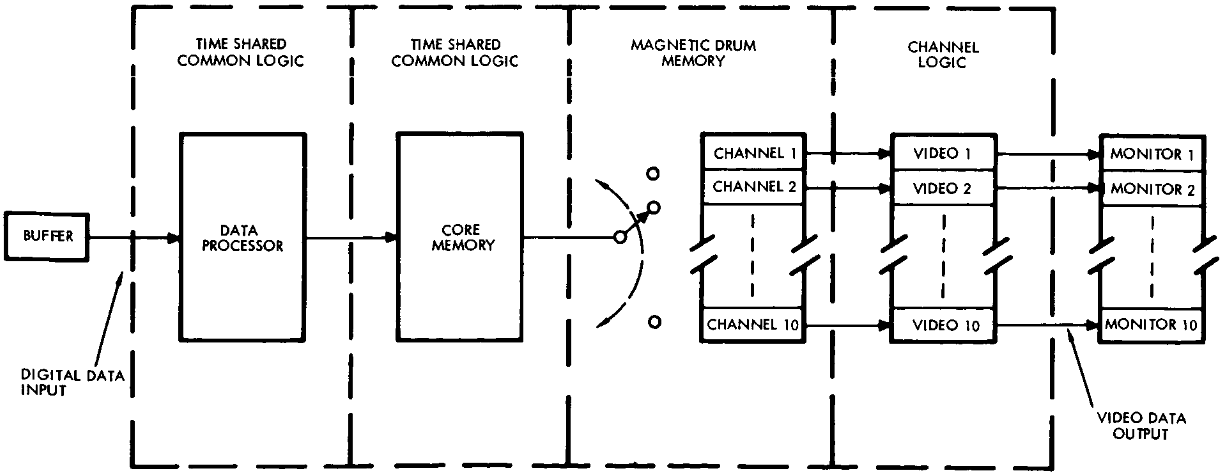


Figure 12-10
OPERATION OF HAZELTINE DIGITAL DISPLAY GENERATOR is shown inside dotted lines. The buffer and the monitors (CRT displays) are not supplied as part of the hazeltine system.

The picture elements stored in the drum memory are indestructable except by deliberate erasure. The drum memory rotates in synchronism with the television frame rate and provides properly timed television video and synchronizing signals for the display monitors on each of 10 channels. The magnetic drum furnishes 16-bit parallel video information; this is converted to normal television serial video in the video processor which supplies either composite or noncomposite video to the monitors.

- The video processor cabinet contains the drum and video generator common logic, drum and video generator channel logic, and the television synchronizer. The major disadvantages of the Hazeltine system are the fact that its cost is significantly greater than the standard computer keyboard entry and display system and the fact that the programming required to service it will be so extensive as to overtax the STC computer. If this is true, a second computer would be required. This second computer could be a smaller computer consisting of elements or the Sigma 2 operating as a satellite computer to the STC computer.

Sigma 5 Consideration. Scientific Data Systems Sigma 5 general purpose computer and associated peripheral equipment is under consideration by TRW Systems for use as the nucleus of the total spacecraft test system. The following presents our rationale for selecting SDS in general and the Sigma 5 general purpose computer in particular to perform the necessary functions required by this system:

- TRW Systems has been using SDS equipment for over four years for all phases of spacecraft testing and launch operations. During this time a great deal of experience and capability on SDS equipment has been developed. This includes programming, systems engineering, and computer operator experience. SDS is located in the vicinity of TRW and conducts training courses for programming and maintenance of their equipment. This proximity facilitates attendance at these courses by TRW personnel assigned to any project using SDS equipment. TRW quality assurance personnel can visit the production facilities and monitor assembly and test operations at no additional cost to the customer. Experience has proven that SDS has capable service engineering personnel and their response time is adequate. SDS maintains a field office at the eastern test range and in the past has provided excellent support during launch operations.

- The Sigma 5 general purpose computer is a high-performance, medium-size computer with nanosecond hardware and a complete line of advanced software. It is ideally suited for the recommended application which requires fast computation with concurrent high-speed input-output. This includes the capability to simultaneously perform general purpose computing in the background, multiple real-time control operations in the foreground, and a large number of concurrent input-output operations.

- A more complete survey and technical evaluation of computers available for this type of application is detailed in TRW IOC 7517.4-2 dated 1 June 1967.

12.2.4 Guidance and Control Subset Functional Description

The guidance and control subset (GCS) provides two paths into the guidance and control system on the spacecraft. First, the GCS provides stimulators which supply stimuli in the form of radiant energy to the sensors aboard the spacecraft. These stimulators operate under direct control of the guidance and control subset. The second path is through the generation of electrical signals which simulate the output of the sensors. These signals are injected directly into the sensor electronics.

The control of the stimulator/simulator output can be directed either manually at the GCS or automatically from the computer. Under computer control only discrete values can be chosen while under manual control the stimulator/simulator outputs are continuously variable. The sequence in which a typical test might be run is as follows:

- Installation Test. A test designed to prevent the wrong application of voltage to each of the separate boxes making up the spacecraft GCS.

- Function Test. A test designed to test each mode of operation including the redundant units and system in the typical flight order using simulators.

- Integrated System Test. A test similar to the functional test but abbreviated. It is run during the time the rest of the spacecraft electrical boxes are being tested.

- Telemetry Calibration. The spacecraft telemetry calibration test will include a calibration of all GCS telemetry.

- GCS Polarity Test. The stimulus units will be used in any logical order to verify the polarity of each sensor. Rates will be induced about each of the spacecraft axes to test the polarity or each of the gyros. Sequencer and antenna position polarity will also be checked.

Equipment Descriptions

A conceptual rack layout is shown in Figure 12-11.

The following is a brief description of the various sensor stimulus and simulator units:

- Canopus Sensor Stimulus. A unit, made small and light enough to be mounted on the spacecraft or the sensor directly, which will contain a collimated light source movable to at least five positions. Provisions will be made for aligning to the sensor and spacecraft axes.

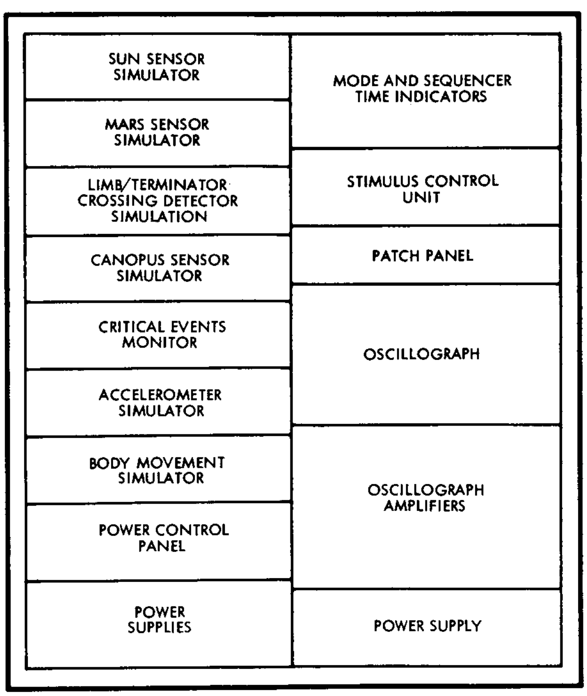


Figure 12-11
GUIDANCE AND CONTROL SUBSET RACK LAYOUT shows the units which comprise the GCS, except for the stimulators which attach to the spacecraft.

- Canopus Sensor Simulator. An electronic unit which will inject electrical signals into the Canopus sensor output lines which will simulate angles of star positions.

- Sun Sensor Stimulator. A high-intensity light source installed on a light frame which will mount on the spacecraft. Provisions will be made for five positions and for aligning to the sensor and spacecraft axes.

- Sun Sensor Simulator. An electronic unit which will inject electrical signals into the coarse and fine sensor simulating various angles of sun positions.

- Limb and Terminator Crossing Sensor Stimulus. An enclosure made small and light enough to be mounted on the planetary scan platform, which will contain a painted disc, half black and half white illuminated by an internal light source. Provisions will be made for aligning to the sensor and spacecraft axes.

- Limb and Terminator Crossing Sensor Simulator. An electronic unit which will inject electrical signals into the sensor simulating a transition of night and day.

- Mars Sensor Stimulator. A heated target mounted within a frame light enough to be mounted on the sensor or the spacecraft above the sensor. The target will have a background large enough to prevent the sensor from viewing stray heated areas. Provisions will be made for aligning to the sensor and spacecraft axes and for changing the size of the target.

- Mars Sensor Simulator. An electronic unit which will inject electrical signals into the sensors simulating a Mars target of various sizes and angles related to the spacecraft axes.

- Body Movement Simulator. Torquer current sources will be provided for each of the three gyros. Each source will be isolated from all other circuits in the GCS EOSE. Each source will provide a variable DC torquer current continuously adjustable over the range required. It also may be required to change the DC current to a current which changes at a constant rate.

- Accelerometer Simulator. An electronic unit which will inject signals into the accelerometer simulating various accelerations of the spacecraft. An automatic sequence of several inputs will simulate a typical flight.

- Engine Position Readout. Engine position will be read out in two axes within the spacecraft and will be wired to the spacecraft interface for oscillograph recording. This will allow comparison of input and output during tests.

- High Gain Antenna and Medium-Gain Antenna Readout. Antenna position will be read out in two axes within the spacecraft and will be wired to the spacecraft interface for oscillograph recording. Comparison of sequencer programming and antenna pointing will be possible during tests.

- Guidance and Control Test Rack. The following will be part of the GCS test rack:

  a. Simulators covered above

  b. Stimulus control units as required covered above

  c. Oscillograph

  d. Patch panel for oscillograph channel selection

  e. Mode and sequence time indicators.

- The critical events monitor has hardline interfaces with the spacecraft to test events in the spacecraft guidance and control subsystem which are not sampled at a high enough rate to obtain the resolution desired during system testing.

12.2.5 Science Equipment Subset

The TRW experience in the integration of scientific experiments has shown that each experimenter will require unique OSE, tailored for his special problems. This OSE normally consists of three portions:

1) Stimulator, which can apply a stimulus to the actual experiment detector elements

2) Simulator, which functionally simulates the experiment detector output

3) Readout, which can monitor, process, and display experiment output data.

This experiment-oriented OSE supports bench functional testing and environmental qualification and flight certification of each experiment.

During system-level assembly and test operations, only the stimulator/simulator portion of the test equipment is required and experiment data is monitored through telemetry and processed by the EOSE data processing subset. The experiment will previously have been tested using the so called experimenters spacecraft simulator, equipment which faithfully simulates the experiment/spacecraft interface. The simulator supplies power, commands, and telemetry inputs to the experiment and outputs data in an actual PCM telemetry format for computer data processing. This then is baseline data, against which system-level test data may be compared.

In order to permit computer control of closed-loop experiment test, TRW will integrate the stimulator (control) and simulator equipment in standard 19-inch relay racks and will interface this GFE equipment with the computer hardware/software.

The photo-imaging subsystem, although provided by TRW, offers the same test problem. The test stimulus in this case will be some form of television test pattern and it is doubtful if a simulator can be used. Photo-imaging data through telemetry will be processed with the digital computer and displayed in a qualitative fashion with relatively simple television monitoring equipment. No television data enhancement equipment is required in the EOSE, although analog magnetic tape data may be processed off-line using existing JPL enhancement equipment.

Because of the stringent contamination requirements on Voyager, it is imperative that experiments and the photo-imaging subsystem be designed with commandable in-flight calibration or similar self-test features. In the case of the photo-imaging subsystem, the field of view in the stored (launch) configuration should be such as to permit use of a test pattern for stimulus, under the shroud.

12.2.6 Interface Simulation Subset Functional Requirements

The need for interface simulators results from the requirement to completely verify and qualify all spacecraft to "other equipment" interfaces. Thus, a capsule simulator is required to functionally simulate the capsule interfaces to enable performance of a planetary vehicle integrated systems test prior to integration of the actual capsule. The use of such interface simulators permits parallel testing in the most cost-effective and timely manner.

This study has identified the requirement for four simulators:

- Capsule simulator
- Spacecraft/launch vehicle simulator
- Propulsion module simulator
- Equipment module simulator.

A spacecraft simulator suitable for DSIF compatibility testing has been considered but rejected in favor of using an actual spacecraft for this task.

Capsule Simulator. This equipment will functionally simulate the capsule-to-spacecraft interfaces for all modes of mission operation; prelaunch, launch, boost cruise, orbit insertion, separation, and capsule entry and descent. These interfaces will include:

- Load simulation for 200 watts of DC power at 36 to 50 volts
- Load simulation and switching for power control commands
- Load simulation and verification for all other commands sent from the spacecraft command and sequencing unit to the capsule sequencing and timing unit
- Load simulation and verification for telemetry control commands sent to the capsule telemetry and data storage system

- Simulated telemetry data in each of three modes:

  1) Very low bit rate data hardlined from the capsule to the spacecraft as in the cruise mode

  2) Low bit rate data (500 bits/sec) UHF radio telemetered from the capsule to the spacecraft as in the entry mode

  3) High bit rate data (100 kb) UHF radio telemetered from the capsule to the spacecraft as in the terminal descent mode.

Spacecraft/Launch Vehicle Simulator. This equipment will functionally simulate the interfaces between the spacecraft and the launch vehicle and the spacecraft and the launch complex. The simulator will be used to verify the integrity of the launch vehicle circuity at the S-IVB IU/Planetary Vehicle Adapter (PVA)/shroud interface and also the facility circuitry at the PVA/shroud interface prior to mate of the planetary vehicles. These interfaces will include:

- Load and source simulation for Planetary Vehicle Adapter instrumentation (both forward and aft PVA's)
- Load simulation and verification for separation signals sent from the IU to the Planetary Vehicle/Planetary Vehicle Adapter separation mechanism (both forward and aft)
- Load simulation for ground power supplied through the shroud umbilical
- Source simulation for telemetry data hardlined through the shroud umbilical
- Source simulation for Planetary Vehicle signals to be monitored through the shroud umbilical
- Load simulation and verification for preflight disconnect activation commands
- Simulated telemetry data S-band radio telemetered from the spacecraft to the launch facility.

In addition, this simulator will include validation equipment for the solar array simulation and power control equipment furnished as part of the hardline/peripheral subset.

Equipment Module Simulator. This equipment will functionally simulate the interfaces between the equipment module and the propulsion module to enable subsystem testing of the propulsion module prior to integration with the planetary vehicle. These interfaces will include:

- Source simulation for power supplied to the propulsion module
- Source simulation for commands from the computer and sequencer or the command decoder
- Source simulation of ordnance firing signals from the pyrotechnic subsystem
- Source simulation of thrust vector control signals from the guidance and control subsystem.

Propulsion Module Simulator. In a complementary fashion, this equipment will functionally simulate the interfaces between the propulsion module and the equipment module and will enable complete testing of the equipment module without the actual propulsion subsystem. The interfaces will be:

- Load simulation for power supplied to the propulsion module
- Load simulation and verification for all commands sent to the propulsion module
- Load simulation and verification for ordnance firing signals
- Source simulation thrust vector control feedback signals
- Source simulation of engineering measurements

Figure 12-14 shows a more detailed definition of the interface between the capsule simulator and the spacecraft. The configuration shown is for both the cruise (mated) mode of test and the separated (entry) mode when only an RF link is maintained.

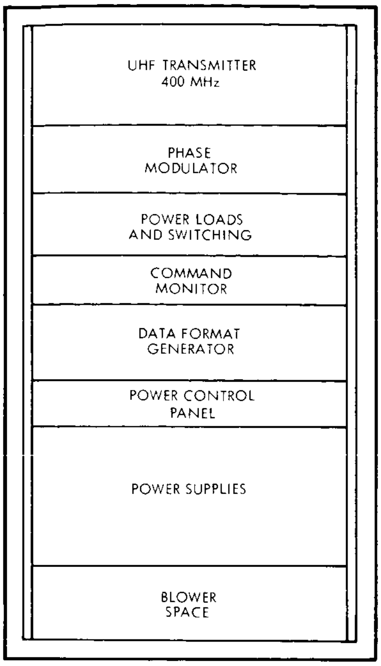


Figure 12-12

CAPSULE SIMULATOR RACK LAYOUT shows the equipment used to simulate the capsule to the spacecraft during testing when the capsule is not available.

Figure 12-15 shows a more detailed definition of the spacecraft/ launch vehicle simulator showing the test configuration required to verify the Planetary Vehicle/Planetary Vehicle Adapter interface (Planetary Vehicle Adapter mated).

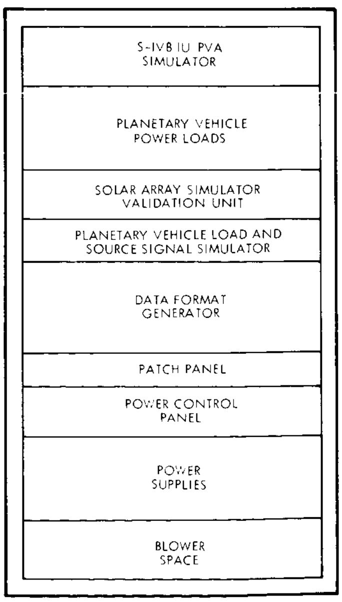


Figure 12-13

SPACECRAFT/LAUNCH VEHICLE SIMULATOR RACK LAYOUT shows the physical relationship of the units simulating the launch vehicle electrically during testing.

Equipment Descriptions

Capsule Simulator. This simulator will consist of electronic drawers, rack-mounted in a standard 19-inch relay rack as shown in Figure 12-12. It will be utilized local to the spacecraft, connected at the capsule umbilical interface through an adapter cable. Individual drawers are as follows:

- UHF Transmitter Drawer. Commercial low power, 400 MHz (approximately) transmitter with necessary power attenuation and monitoring capability.
- Modulator Drawer. TRW-designed phase modulator to permit direct modulation of simulated, low and high bit rate PCM telemetry data on UHF carrier, to include modulation index controls and monitoring capability.
- Power Loads and Switching Drawer. TRW-designed drawer containing 200-watt load bank, power transfer relay, and status indicator.
- Command Monitor Drawer. TRW-designed drawer containing load simulation and status indicators for all discrete commands and a command decoder to accept, decode, and display all serial commands.
- Data Format Generator. Commercial PCM data simulator with flexible capability for PCM code, bit rate, word, and frame format. This will probably be the same unit used for self-test of the data processing subset.
- Power Control Panel. 115-volt, 60 Hz AC power control circuit breakers and running time indicators.

Spacecraft/Launch Vehicle Simulator. This simulator will consist of electronic drawers, either rack-mounted in a single enclosure as shown in Figure 12-13, or individually mounted in portable suitcases for ease in use on stand. The equipment will be capable of testing the shroud to Planetary Vehicle Adapter interface (prior to Planetary Vehicle Adapter mate) or at the Planetary Vehicle/Planetary Vehicle Adapter interface (after Planetary Vehicle Adapter mate). Individual drawers are as follows:

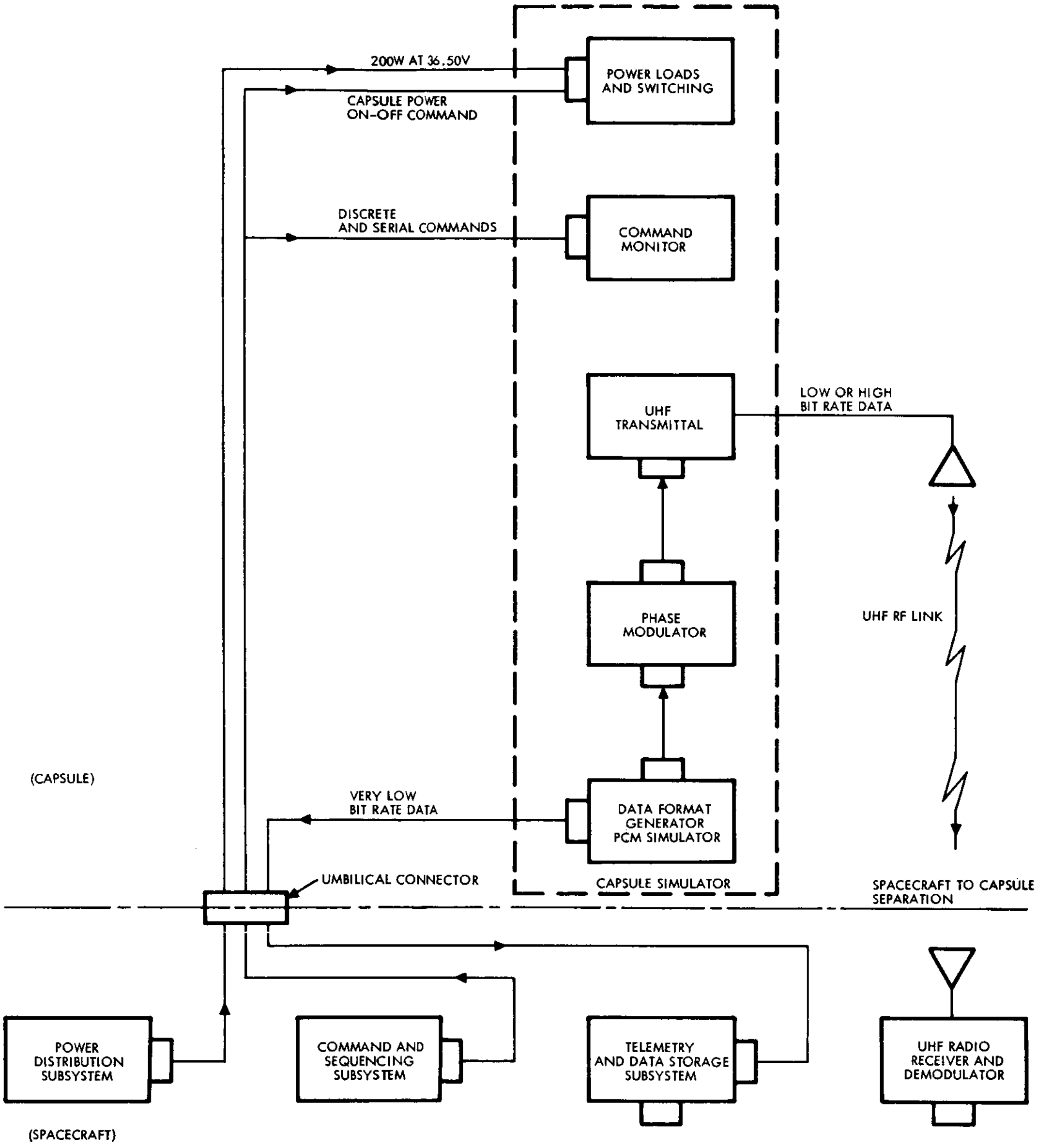


Figure 12-14

CAPSULE SIMULATOR ELECTRICAL INTERFACES show the signal lines from the spacecraft to the equipment which simulates the capsule during testing.

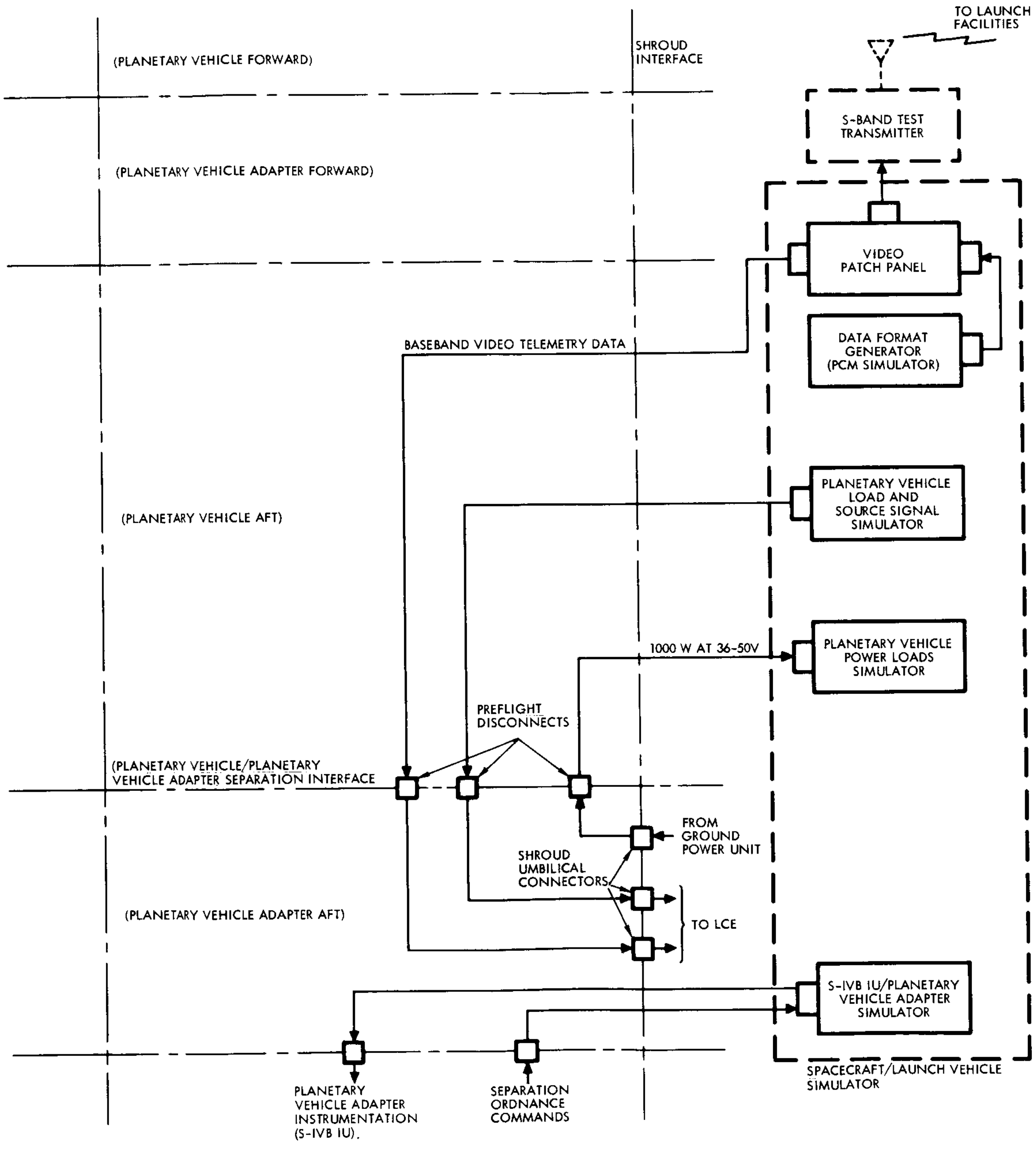


Figure 12-15
SPACECRAFT/LAUNCH VEHICLE ELECTRICAL INTERFACES are accomplished by signal lines from the spacecraft to vehicle during testing.

- SIV-B IU/Planetary Vehicle Adapter Simulator Drawer. TRW-designed drawer containing Planetary Vehicle/Planetary Vehicle Adapter separation ordnance simulation and monitors and simulated Planetary Vehicle Adapter instrumentation source simulation. Both fore and aft Planetary Vehicle Adapter's are simulated. This drawer can also be used for instrument unit checkout prior to mate with the SIV-B.

- Planetary Vehicle Power Loads Drawer. TRW-designed drawer containing load banks and metering.

- Solar Array Simulator Validation Unit. TRW-designed drawer used in conjunction with Planetary Vehicle power loads drawer to setup, calibrate, and validate the solar array simulator and power control portion of the hardline/peripheral subset.

- Planetary Vehicle Load and Source Signal Simulator Drawer. TRW-designed drawer containing simulation of all functions monitored or controlled through the shroud umbilical.

- Data Format Generator. Commercial PCM data simulator with flexible capability for PCM code, bit-rate, word, and frame format, will simulate hardlined telemetry data monitored through the shroud umbilical or can be used to modulate a test transmitter (not furnished).

- Patch Panel. Video patch panel for distribution of simulated telemetry data.

- Power Control Panel. 115-volt, 60-Hz AC power control panel circuit breakers and running time indicators.

- Power Supplies. If required.

Equipment Module Simulator. This equipment will consist of one electronic drawer, packaged in a portable suitcase for ease in use at remote propulsion test facilities and will contain all loading and source simulation together with integral power supplies.

Propulsion Module Simulator. This simulator will consist of one electronic drawer, packaged in a suitcase, and will contain all necessary load and source simulation.

12.3 SYSTEMS TEST COMPLEX SOFTWARE

Two programming packages will be supplied with the STC. One package will consist of the standard programs available from the computer vendor. The other package will consist of all the programs required for performing system tests on the spacecraft. This second group of programs performs data selection, decommutation, reduction, analyses, display, and logging functions.

All the programs will be contained in object program relocatable format on a magnetic system tape along with a program for transfer of programs from magnetic tape to generate the disc-resident operating program library. Programs required may then be called into the computer from the disc and executed.

Program documentation will include operations manuals, program listings, program flowcharts where applicable, data interface descriptions, card and record format descriptions, program theory of operation and restrictions. Where practical, program documentation will be contained in program listings which will be distributed on magnetic tape.

12.3.1 Standard Vendor Software

The programs and subroutines which will be produced by the computer vendor are listed below. These programs will be used by TRW in the development and checkout of the spacecraft system test programs. Many vendor-furnished programs will be used directly to support spacecraft integration and test activities. Where necessary, TRW may modify vendor-furnished software to meet specific requirements.

- Mnemonic Assembler. This program generates absolute and relocatable object code in machine language using symbolically coded source program card images for input. All computer instructions are accepted by the assembler. Memory addresses are expressed in symbolic, decimal, or octal representation. Arithmetic or logical operations may be symbolically indicated; the assembler will perform required operations to compute absolute or relocatable addresses. The assembler has provisions for symbol definition external to the program to permit program segments assembled separately to interface data and subprograms. The assembler program will be extensively used in the development and modification of test and test support programs.

- Fortran IV Compiler. This compiler will include as a subset the American Standards Association USASI Fortran IV. This standard is substantially identical to IBM 360 Fortran IV. The compiler will permit the use of assembly-language program segments at load time to permit use of Fortran programs for low data rate processing in real time. The Fortran compiler has provisions for selective tracing usable for program checkout. It also has provisions for chain loading of program overlay segments to permit execution of programs larger than the core memory of the computer. Output of the compiler is a source language listing which includes a memory map giving common assignments, array allocations, subroutine names, and variable assignments. The compiler will provide a high degree of syntax analysis to detect inconsistent use of symbols and the detection of programming errors in the source program.

  The Fortran compiler will be used for small engineering problems such as geometric calculations used in sensor alignment procedures. Test procedure documents and inventory/spares packing lists can be tabulated using Fortran IV programs for printer formating from cards.

- Program Loaders. A number of loader programs will be used to input absolute or relocatable binary object programs into the computer's core memory. Binary input may be from card reader, paper tape reader, magnetic tape, or disc. The system loader contains the capability to select programs by ID name, to load library relocatable subroutines to satisfy external references, to output a memory map of program and array locations, and to overlay program instructions when chain-linked execution is required. The loaders will use standard binary formats produced by the mnemonic assembler, the Fortran IV compiler and other compilers.

- Subroutine Libraries. A number of general purpose subprograms will be furnished in relocatable binary form to make up a subroutine library. The library includes the following:

  1) Fortran run-time system subroutines

  2) Mathematic functions-Fortran IV math library including trigonometric, logarithmic, exponential, and hyperbolic functions. Integer, floating point, double precision, and complex arithmetic operations are included.

3) Format conversion subroutines: binary to BCD, BCD to binary, floating point to fixed point, etc.

4) High-Performance real-time peripheral driver subroutines for printer, CRT displays, disc, magnetic tape, card reader/punch, typewriter/keyboards. These routines permit multiple concurrent input/output operations without delaying central processor computation except for setup of external input-output control hardware registers.

5) Incremental plotter subroutines for control of plotting for generation of scales and for annotation. These routines will permit the automatic generation of trend graphs and calibration curves.

To the extent practical, considering the need for high-speed computation in real-time testing, these library subroutines will be used in programs to support test activities and in the analysis and reduction of test data. In some cases, essentially equivalent math functions may have to be specially coded to optimize computing speed for critical real-time requirements.

- Library Maintenance Program. This program facilitates the addition, deletion, and replacement of programs onto the program library medium. Entire programs can be added to the library media (tape or disc file) under an ID name for loader reference. The library maintenance program will also assist in required changes to the subroutine library file.

  Program libraries will normally reside on magnetic tape for backup and for distribution purposes. A program will be provided to copy a program library onto disc to generate a disc operating system.

- Aids to Program Checkout. A number of programs will be used to facilitate checkout of computer programs under development. Features included are memory or media dumps, memory searches, memory snapshot dumps during program operation, and keyboard input alterations to memory. These programs permit programmers to inspect computational results and computer instructions or to make changes to program logic during program testing and program validation. They are also useful in distinguishing between program errors and hard to isolate hardware malfunctions.

- Media-to-Media Conversion. These routines will be provided to facilitate the transfer of data from one storage medium to another, e.g., card to tape, tape to disc, disc to printer. Variable-length record data transfers between any two of the following devices can be made using these routines: magnetic tape, disc file, card reader/punch, teletypewriter, paper tape reader/punch, line printer, display/keyboard.

  A program will be provided to permit editing of source language data contained on magnetic tape or disc. This program will be used to insert, replace, or delete card(s) prior to the compilation, assembly, or listing of the source data. In this manner the computer can be used to keep large source program files and textual data files up to date.

- Diagnostic Routines. These routines provide the capability for exercising the computer and all vendor-furnished peripheral devices. Diagnostics will facilitate isolating specific hardware malfunctions. The central processor unit (CPU) is tested by executing all instructions with worst-case data and comparing results with known correct results. Memory, storage media, and input-output channels are tested with worst-case and random data. All permissible data transfer modes with the peripheral input-output units are tested. Detected errors result in visual indications, program halts, or printed messages. Diagnostic programs permit operator control for the selection of test options to permit program recycling in regions of detected malfunctions to facilitate signal and pulse analysis.

  Vendor-furnished diagnostic programs will be augmented with additional programs to make up a validation program package which can be used to verify correct operation of computer-connected data buffers, display panels, and special-purpose peripheral devices.

- Monitor Programs. Two monitor programs will be furnished by the computer vendor - a batch processing monitor and a real-time monitor.

1) The batch processing monitor permits efficient use of the computer for program assemblies, compilations, media conversions, and execution of non-real-time post test data analysis programs. The monitor permits swift sequencing from one job to the next and allocates the use of peripheral data storage media.

2) The real-time monitor program in conjunction with the disc operating system library permits the computer to service displays and printer in real time. It has facilities for program call-up and loading based on requirements initiated by interrupt (foreground) processing subroutines. The real-time monitor schedules computational, memory, and disc resources of the system in the real-time monitoring of the low-rate engineering spacecraft data. To the extent that system resources are not utilized for testing (during standby periods), the monitor may allocate the unused capacity of the computer for other tasks such as program assemblies or procedure generation.

12.3.2 Spacecraft Test Programs (Real-Time)

TRW Systems will develop all the programs required for spacecraft and capsule testing. This section describes the characteristics of the real-time test support computer program system used during functional tests, environmental instrumentation modules.

Prior to integration, spacecraft modules may also be bench-tested using a hardware simulator for the spacecraft digital data handling systems. Thus, the telemetry data from the bench-tested module will also be compatible in format with the telemetry input subprogram in the real-time test support program. The real-time test support program can be used to support bench tests without interfering with spacecraft integration activities.

The real-time test support program system consists of several organic subprogram modules. They are tabulated below:

- Executive program task scheduler
- Telemetry input routines - high and low bit-rate links
- Discrete status monitor routines for spacecraft and STC equipment

- Data routing and editing routines
- Data processing routines
- Data recording routines
- Data display routines
- Command generation and verification routines
- Stimulus monitoring and controlling routines
- Input-output device control subroutines
- Program option input routines.

Executive Program Task Scheduler. This program will operate concurrently with the interrupt level (foreground) subprograms, but with lower priority (background). This program will call the required subprograms to perform required processing. The executive program receives telemetry status from the interrupt processing subroutines, receives control information from option input routines, and receives automatic requests from peripheral devices for data. The executive allocates computer processing resources to data routing and data processing, keeping the data buffers filled and up to date. In case a user request calls for a program or for a data block located on the disc memory, the executive schedules its retrieval and schedules subsequent processing when it arrives in core. The executive program will call vendor-furnished routines as well as the TRW Systems routines discussed below to carry out the detailed data processing required for real-time spacecraft test support.

Telemetry Input Routines. These routines operate under interrupt control and communicate with the executive scheduler. These routines control the input of either the engineering data (low bit rate) or scientific data (high bit rate) or both concurrently. The interrupt routines control the appropriate computer memory input channel to permit data to automatically enter the specified memory buffer.

Since the decommutation equipment can input data changes, the telemetry input routine will notify the executive routine of any need for data processing of this changed data for display updating. When the decommutation equipment flags an out of limits data condition, the telemetry input routines signal the executive and transmit revised limit values to the external comparators.

The telemetry input routines continuously monitor synch status, frame synch, and subcommutator synch, as a check on data validity. Word parity error counts are kept for use for error rate calculations. Data with bad parity or data received during intervals of synch loss is screened from further processing.

Discrete Status Monitor Routines. The telemetry routines monitor data mode and format status bits to determine the structure of the data stream. Frame composition models, subcommutation models, and supercommutation models are consulted in sorting out the data for movement into appropriate secondary telemetry buffers where data placement is independent of data mode. Mode status information and coding classification, and unencoded or decoded biorthogonal data information are communicated to other program routines via a discrete status block of memory.

Status of STC equipment and stimulus generating equipment is also monitored, and this status information is also stored in the discrete status block. These routines may be controlled by the executive program to periodically sample slowly changing inputs. Inputs which can change more rapidly may be tested in telemetry interrupt routines.

Data Routing and Editing Routines. Some data routing and editing is performed on telemetry data by the telemetry input routines as described above. After the secondary telemetry buffers have been filled, their contents will be used by other routines under executive control. Some of the associated routing functions are listed below:

- Route to magnetic tape for post-test off-line processing
- Route to disc to buffer high peak-rate data with low duty cycle
- Select data for scientific instrumentation module routines

- Select time-tagged commands for verification from dumps

- Select data for processing to produce displayed quantities

Data Processing Routines. These routines comprise many processors. One processor is used to convert new telemetry to formated alphanumeric data and to discretes for the CRT and status monitor display. As telemetry data or external discrete status changes are noted by the computer, the corresponding display data blocks are updated. Another processor is used to provide status snapshot reports on the printer during the course of a test. A third processor logs pertinent data on disc to form test history blocks. A fourth transfers display data to printer.

- Display Processing. Display processing programs convert raw telemetry data to meaningful terms - to engineering units or to discrete event (on or off) status. Then this data is tabulated in display data subset blocks for output to displays at the CRT display user's request. The display user may select a group of up to 30 words or he may select any of a number of data display subset blocks depending on his needs:

  1) Electronic power subsystem data
  2) Temperature data
  3) Propulsion system data
  4) Guidance and attitude control system data
  5) Deployment status data
  6) S-band radar monitor data
  7) Data handling system status
  8) Spacecraft status critical values
  9) Scientific instrumentation status displays
  10) Command sequence displays - to be transmitted or time-tagged command storage
  11) External EOSE stimulus status data
  12) Computer data processing/routing status
  13) Requested data word blocks (up to 30 words).

Warning conditions for data will be flagged by use of a dedicated portion of the display area. Warning conditions might be triggered by out-of-limits data or by high rate of change of critical data. Provision will be made for warning acknowledgement and consignment display suppression. Subsequent to display suppression, if new data indicates a worsening condition, suppression override criteria might call for a new warning. Specific conditions which will trigger warnings:

1) Data has changed from in-limits to out-of-limits

2) Data has changed from out-of-limits to in-limits

3) Data change has exceeded a predetermined delta

4) Data rate of change has exceeded a predetermined threshold.

- Printer Test Status Logging Snapshots. On command of the test personnel, a snapshot of the status of the spacecraft engineering data and the capsule scientific instrumentation status will be output on the printer. Processing routines similar to those used to convert data for displays will convert all or selected subsets of the data for reports. As with the displays, data may be grouped by related functions on pages of the snapshot reports.

- Test History Data Blocks. Whenver a snapshot is performed, and automatically as displays are updated, a test history block for selected data will be updated. This block will contain average, maximum, and minimum values in engineering units for selected measured values. Status information may also be present. Over long tests, these history blocks will be transferred to disc for storage. At the end of the test or on command, this test history file can be transferred to digital magnetic tape. Later, data on this and other similar history tapes may be computer-correlated to produce trend analysis data off-line.

- Printout of Display Data. Whenever a display operator requires it, the present contents of his display can be routed to the printer and output. If the printer is currently in use he will have to wait for it to be released.

Data Recording Routines. When it is necessary to record all telemetry data during a test, it will be done outside the computer system by recording the PCM bit streams on wideband magnetic tape recorders. Reversed playback telemetry data will be recorded in this manner. Data will be played back (re-reversed) into the computer for processing.

The computer program will also have data recording capabilities; both high and low bit rate PCM data can be recorded. Data can be edited by the computer in real time to compress the data. The digital tape will be optimally blocked nine-track IBM compatible digital tape.

Data reduction editing will be based on the following criteria:

- Pretest and post-test data can be omitted
- Data recorded can be only on change basis
- Data may be selected by telemetry word assignments
- Data input during synch loss will be omitted
- STC equipment status and test condition data may be recorded.

The data recording routines control the filling of the magtape write buffers. If errors are detected, to the extent possible by tape timing restrictions, the data recording routines will erase and write the data on a new area of the tape. High peak volume data may be buffered onto disc in burst modes of data recording. Later data can be transferred from disc to tape at lower average data rates.

Data CRT Display Routines. These routines control the CRT displays and also control discrete display indicators in the STC. Several levels of program control are necessary. Data input to these routines is in the form of binary coded alphanumeric data subset blocks which may include alphanumeric status information. The subset block has space reserved for warning condition messages and space for interconsole communication purposes. In addition to alphanumeric text, there are codes reserved for limited special-purpose graphics symbols such as bar graphs and line-point graphs.

The routines in the main computer transmit data to a satellite computer where data is combined with formating information to produce the required display. In the reverse direction when a display user wishes a printout, the alphanumeric data is moved from the data subset block to printer buffers for output. If computing duty-cycle permits, the satellite computer may also be used for other output data conversion purposes.

As changes in data are detected, the telemetry input buffers are updated; the appropriate data subset blocks are updated; the updated information is sent to the satellite computer; and, finally, the display itself is modified.

Command Generation and Verification Routines. These routines will control the generation of all commands through the computer. Manual requests from the test conductors console will allow single commands to be selected. Preprogrammed command sequences from cards or from disc may also be initiated from the test conductors console display keyboard.

All commands will be checked for permissibility prior to transfer to the command generator. If a requested command is not in the permissible list, a warning message will be routed to the test conductors display and an indicator lamp will be lit on the test conductors console. The command will not be sent.

If a requested command is permissible it will be sent to the spacecraft through the command generator. While the command is being transmitted it will be checked bit by bit by a verification subroutine. If an error is detected, transmission will be stopped and an error warning and light will be displayed on the test conductors display.

The spacecraft telemetry will be monitored for direct telemetry indication of receipt of the command (during testing the interval is short). Comparisons will be made with command transmitted and command received. Again discrepancies will be reported to the test conductor via warning displays.

If all goes well, the test monitor computer will adjust appropriate telemetry screening limits as required to detect abnormal behavior of the spacecraft. Then the command generation routine will seek the next command, manual or in sequence.

Control options will be provided to the test conductor to display the next command in sequence, to step forward or back without transmission, and to select alternative command sequences through the card reader.

Stimulus Monitoring and Controlling Routines. Closed-loop testing will be a feature of the STC hardware and software. Two types of closed loop testing will be used:

- Computer generates stimulus signals and monitors spacecraft response
- Computer monitors external stimulus and monitors spacecraft response.

In both cases the computer program can compare expected response to actual response and can flag discrepancies to test personnel. Closed-loop techniques are useful where large numbers of operations need to be tested rapidly or repetitively. The computer is an ideal vehicle for executing a programmed series of test steps.

When performing closed loop tests the Executive Control Program schedules stimulus and spacecraft command generation. The stimulus monitoring routines and controlling routines communicate parameters to response prediction routines. Predicted responses are compared with incoming real-time telemetry.

In the case of externally generated stimulus, the status of the stimulus can be recorded along with spacecraft telemetry. Detailed analysis of response can be performed from recordings in non-real time.

Input-Output Device Control Subroutines. These routines are used to transmit control signals to computer peripheral input or output devices, buffers, and registers. These routines sense error indications flagging faulty data transmission. When excessive errors are encountered the executive is signalled. Normally only a limited number of automatic transmission re-tries are necessary for successful data transmissions between units.

The device control subroutines operate under interrupt control to optimize data transmission rates. When data is to be transmitted or received and a unit is available for use, these routines communicate data buffer locations to the external device and indicate data block sizes involved.

Program Option Input Routines. Human operators ultimately control both the test and the computer. Although many parts of tests may be pre-programmed to operate automatically, unexpected events require human decisions and evaluation.

Control options select routines, evaluate computer status, and compare signals from peripheral display keyboards and control panels. Card input is also evaluated to determine whether changes must be made to the program mode tables which direct the executive program task scheduler.

Operator control inputs influence data input, editing, routing, recording, processing, display and printing. The operator controls stimulus generation and command transmissions to the spacecraft. The operator is able to change limit tables, change display contents, change calibration coefficients, and modify tables of expected response to stimulus.

12.3.3 Test Support Programs (Non-Real Time)

The computer system with its card reader, magnetic tapes, printer, and disc provides a powerful general purpose data processing system. While real-time test support is in process, all peripherals are committed to test support activities. During standby periods and after real-time testing is completed, the computer is available for a host of test support activities:

- Instrumentation calibration and limits table revision
- Program reassembly and checkout
- Data reduction and analysis from PCM and digital magnetic tapes
- Text editing and printing
- Data history reporting and trend analysis
- Fortran engineering calculations.

Instrumentation Calibration and Limits Table Revision. When the computer is processing data in real time it uses numerous tables for conversion of raw telemetry to engineering units, for data limits specification, for assignment of telemetry words in different data modes, for modeling subsystem responses to test stimulus, or to different spacecraft command states. These tables will reside on the disc.

It is often necessary to change the contents of these tables to reflect changing spacecraft configuration in the various stages of integration. Non-real-time programs are used to update and revise such calibration and instrumentation data in preparation for new tests. After calibration tables have been revised, a calibration book program will be used to tabulate the engineering units, limits, etc., as a function of telemetry counts. With the use of plotter subprograms, graphs showing calibration relationships can be automatically generated on magnetic tape for plotting off-line.

Program Reassembly and Checkout. Many months of program development must be completed before the real-time and support programs are ready for use. Revisions and improvements will continue to be made to these programs to tailor their features to new test applications. The computer system will be used in non-real-time for program assembly and checkout. Tape recorded spacecraft and simulator data will be used to verify compatible operation of the real-time program with the STC equipment.

Data Reduction and Analysis from PCM and Digital Magnetic Tapes. Operation from PCM tapes simulates real-time operations. The real-time test support program will be used.

Data processing from digital magnetic tapes can be computation-speed limited with no loss of data. High-bit-rate telemetry data can be processed with more complex analysis than is practical in real time. The printer will be used to produce tabular reports of information of significance. Since digital tapes are IBM-compatible, they may be used as input for other computers at TRW or elsewhere.

Text Editing and Printing. A number of test-related documents will require frequent revisions. Many documents such as test procedures, spare-parts inventory lists, or packing checklists can be produced on IBM cards. Large documents can be stored on magnetic tape, edited by the computer, and printed on reproduction mats at high speed.

The calibration data, limit tables, etc., will be edited in the same manner, using the same programs. Considerable clerical costs will be saved by the reduction of human errors and elimination of unnecessary proofreading.

Data History Reporting and Trend Analysis. The real-time program produces magnetic tapes with test history data, minima, maxima, and average values for selected spacecraft data. Data on these tapes will be combined to produce a master test history file. Condensed averages can be extracted from this file of past test results for all tests where a particular unit and associated measurement were recorded. The tabulated reports produced show a history of a single measured quantity throughout the entire test sequence. Hours of manual, error-prone data copying is eliminated. Trend graphs can be produced if required by feeding numeric output to the plotter subprograms to generate plot tapes for plotting on off-line equipment.

Fortran Engineering Calculations. The Fortran IV system of programs is easy to use for small- and medium-scale scientific and engineering computations. Such calculations arise wherever repetitive or iterative solutions are needed. Geometric calculations for sensor alignment is one example. Testing of mathematical models and curve fitting for calibration measurements are frequently required.

13. EOSE/MDE COMMONALITY

Since many of the tasks of the EOSE are similar to those of the MDE, some equipment can be identical and interchangeable. The similarities fall mainly in the telemetry processing and command generation equipment. Several items of equipment have been identified which will be identical between MDE and EOSE. These items are the command encoder, video reconstruction equipment, demodulator/synchronizers, and certain test equipment such as the data format generator. In addition, the EOSE will use the biorthogonal decoder developed for the MDE. The biorthogonal decoder will, however, be developed primarily for use in MDE as opposed to the other items which will be developed for both MDE and EOSE. There will naturally be differences in specifications for EOSE versus MDE, but for cost effectiveness in the cases of common equipment it will be built as a single design to the more stringent specification.

Finally, TRW feels that it would be highly desirable to have commonality or, at least, program compatibility between the STC computer and the computers installed at the DSN stations. Preliminary indications are that a third-generation computer will be installed at the DSN in advance of the Voyager mission. Thus, the possibility of compatibility between the STC computers and the DSN computers (specifically the telemetry and command processor computers) will be enhanced.

14. MOSE INTRODUCTION

Design of the MOSE began with an analysis of operational flows to derive functional requirements. These functional requirements were then consolidated into end-item requirements so that conceptual designs could be developed. End-item requirements were also correlated to time-phased operational flows for determining quantities, need dates, and allocations.

The functional analysis conducted for this study primarily dealt with spacecraft assembly and checkout, system test, and launch operations. Brief functional descriptions are included here for all major items of MOSE. Schedule and implementation information for the MOSE has been integrated with that for the EOSE in Section 2.3 of Volume 8.

Section 15 of this volume contains a pictorial flow showing the use of MOSE.

The MOSE design objectives are presented in Section 16.1. Section 16.3 presents the criteria and constraints considered for the MOSE. It also includes a table of applicable documents. Section 17 contains end-item functional descriptions and a list of required MOSE.

The system and subsystem functional descriptions include specific functional design requirements, test requirements, interface definitions, general equipment descriptions, and a preliminary design for each major item of MOSE.

The dimensions shown in the functional descriptions are derived from past experience with similar items, not from analysis, and are preliminary. Because of their simplicity, functional descriptions for slings are not included. However, slings are shown pictorially in the functional descriptions of the sling-interfacing MOSE.

15. MOSE FUNCTIONAL FLOW

The following two pictorial flows (Figures 15-1 and 15-2) give an indication of the MOSE usage at TRW and Kennedy Space Center (KSC). The sequence of operations was derived from analysis of the assembly, test, and integration flow, and launch site operations flow.

These pictorial flows do not indicate every operational procedure. They are meant to highlight the movements of the spacecraft and major spacecraft tests.

The spacecraft is maintained in a clean condition at all times, including during intraplant handling, transportation, and launch operations. Three types of transporters are required: a bogie for intraplant movement, a spacecraft transporter, and a transporter for the encapsulated planetary vehicle at KSC.

At TRW, major in-plant handling is required for inverting and placing the spacecraft into the thermal-vacuum chamber. Stacking for the vibration and acoustic test and performing the weight and center-of-gravity test are also extensive handling procedures. At KSC, the major handling operation is the buildup of the planetary vehicle.

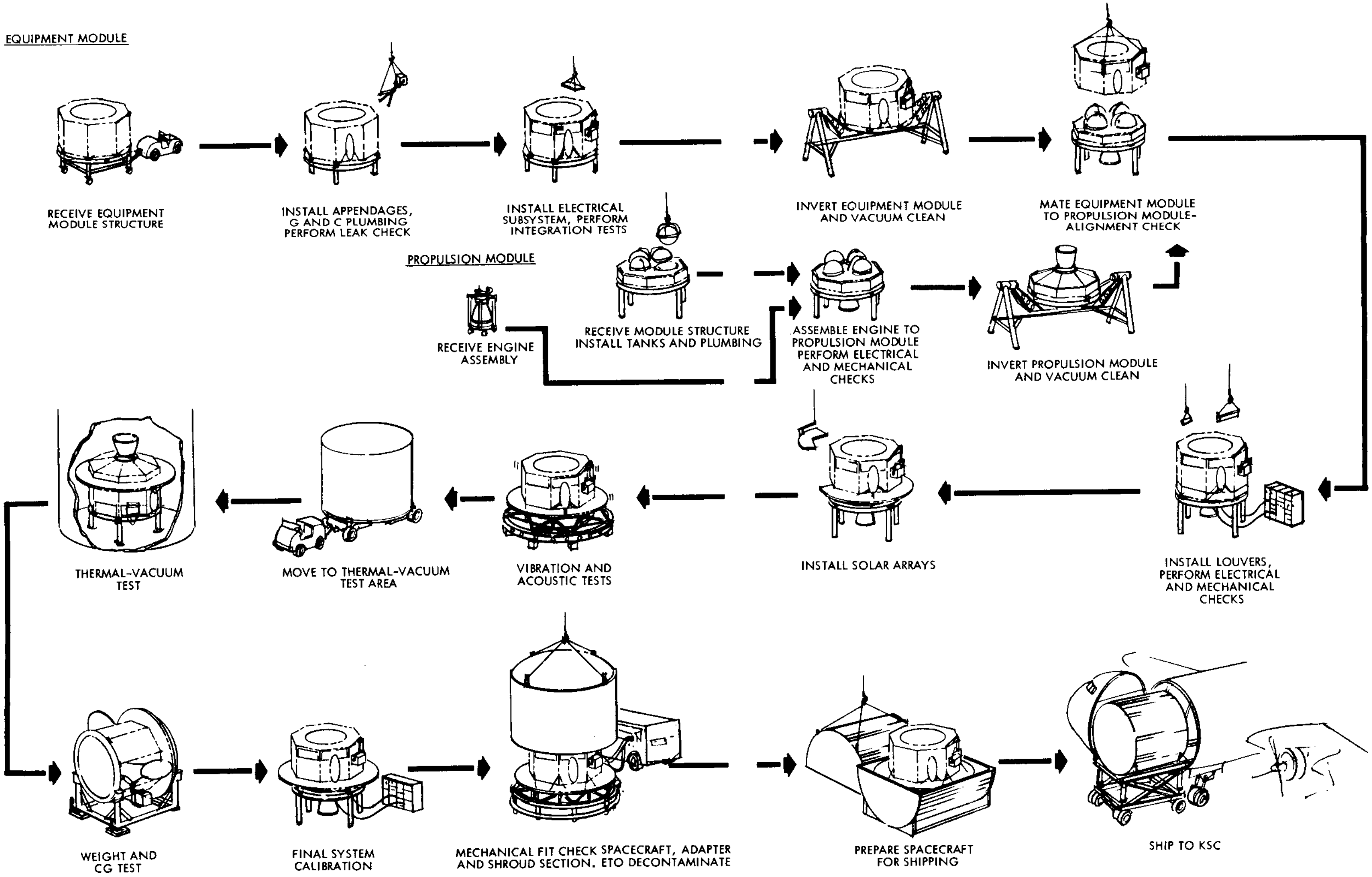


Figure 15-1
MOSE USAGE AT TRW. The vibration, thermal, and weight test require major spacecraft handling operations.

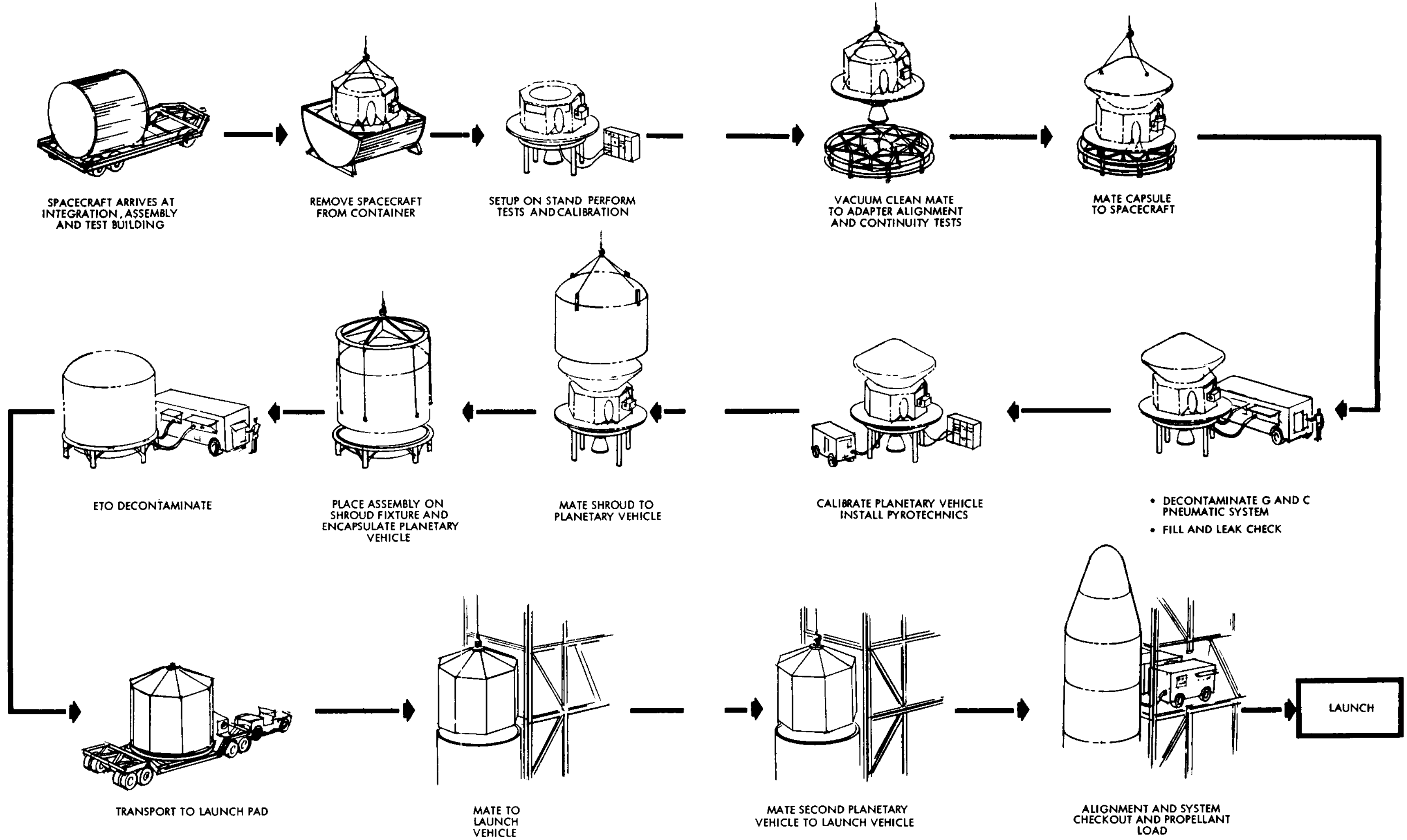


Figure 15-2
MOSE USAGE AT KSC. Stacking transporting and hoisting are the major planetary vehicle handling operations.

16. MOSE OBJECTIVES

MOSE is that mechanical equipment used to lift, hold, position, align, and transport the spacecraft and its major components. It also includes equipment to cool, provide a conditioned environment during transportation, and to decontaminate and sterilize the spacecraft. It provides a pneumatic test console to test the attitude control system. It also provides the fluid handling and control equipment for support of the propulsion system.

16.1 GENERAL OBJECTIVES

a) Handling fixtures are provided for items which are either too large, too fragile, or too heavy for normal personnel handling. Where precise mating of parts is required, especially when the parts are heavy, a precision slow rate hoist attachment is provided.

b) All MOSE is designed for use by experienced technicians. The equipment is compatible with the operational flow for assembly and checkout, spacecraft test, and launch operations of the spacecraft, and with the development test and manufacturing flow of subsystems.

c) Whenever possible, ground support equipment developed for other programs, such as Apollo and LM, is utilized without modifications, even if it has excess capabilities. When modifications are required, they are designed to permit easy retromod. GFE items are noted in Tables 17-1 and 17-2. The design of new MOSE for Voyager-peculiar requirements has been based on experience and capability developed in other TRW programs.

d) All load-carrying MOSE will be proof-loaded prior to interface with end items and again periodically to verify capability. Operating MOSE will be functionally tested prior to initial use and again periodically to verify proper operation.

e) All test and assembly procedures will be verified with the engineering model before they are used with proof test and flight spacecraft.

16.2 DESIGN OBJECTIVES

Multiple-use features are incorporated in the design of MOSE whereever possible and feasible. Individual support equipment units will be capable of servicing different spacecraft and components of the same design, without requiring recalibration or modification of the support units.

All MOSE for the Voyager spacecraft has the following basic characteristics: weight and size are as necessary to service the spacecraft and its parts (without constraining spacecraft design in any way); design is simple and has complete compatibility (material, functional, and magnetic) with the spacecraft; and stable adjustment and positioning provisions are included to eliminate readjustment during tests.

Reliability is ensured by standard components, proven design concepts, and conservative design approaches providing easy operation with minimum maintenance. Maintenance can be performed in a safe and comfortable fashion, using standard hand tools. Access is provided for repair and replacement, test, inspection, fabrication, and assembly.

16.3 CRITERIA AND CONSTRAINTS

The following general design criteria apply to all MOSE; applicable documents are listed in Table 16-1. Item-peculiar design criteria are listed in the functional descriptions of the individual items.

16.3.1 Safety Standards

Safe operating conditions will prevail for both the equipment being handled and the personnel involved. Specifically, the design of MOSE will conform to the requirements of the General Range Safety Manual, AFETR M-127-1. Safety considerations cover interfaces between the MOSE and the operator and between the MOSE and the spacecraft.

16.3.2 Personnel Factors

The probability of operator error is kept to a minimum through detailed and logical procedures, clearly visible instructions and caution plaques, sufficient working space, and readily accessible and comfortably operable controls. Noise levels from operating equipment are kept low to permit unambiguous voice commands. Where special hazards

Table 16-1. MOSE Applicable Documents*

TRW Systems Process Specification	PR 8-1
Federal Standard 209	"Clean Room and Work Station Requirements, Controlled Environment"
MIL-HDBK-5	Safety Margins
MIL-A-8421	Ultimate Load Factors
MIL-STD-129	"Marketing for Shipment and Storage"
MS-33586	"Metals, Definition of Dissimilar"
MIL-D-3716-A, Amend. 2	"Desiccants, Activated, for Dynamic Dehumidification"
MIL-E-5556-A, Amend. 1	"Enamel, Camouflage, Quick Dry"
MIL-M-008090	"Mobility Requirements, Ground Support Equipment, General Specification for"
MIL-C-13777-D, Amend. 1	"Cable, Special Purpose, Electrical, General Specification for"
MSFC-SPEC-164	"Cleanliness of Components for Use in Oxygen, Fuel, and Pneumatics Systems, Specification for"
PPP-B-621-A, Amend. 2	"Box, Wood, Nailed and Lock Corner"
MIL-D-3464-B	"Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification"
MIL-C-9959, Amend. 1	"Container, Flexible, Reusable, Water-Vapor Proof"
MIL-B-26195-A	"Boxes, Wood Cleated, Skidded, Load Bearing Base"

*MOSE applicable documents - utilized for specifications or as guidelines for MOSE design.

Table 16-1. MOSE Applicable Documents* (Continued)

PPP-B-601-A, Amend. 2	"Boxes, Wood, Cleated, Plywood"
MIL-P-116-D	"Preservation, Methods of"
PPP-B-636-C	"Box, Fiberboard"
MIL-STD-803 A-1	Human Engineering Design Criteria for Aerospace Systems and Equipment
MIL-P-9024-B and C	"Packaging, Air Weapons Systems, Specifications and General Design Requirements for"
MIL-STD-1186	"Cushioning, Anchoring, Bracing, Blocking, and Water-Proofing, with Appropriate Test Methods"
ICC Tariff No. 15	"Regulation for Transportation of Explosives and Other Dangerous Articles"
Air Force Manual 71-4	"Packaging and Handling of Dangerous Material for Military Aircraft"
MIL-B-131	"Barrier Material, Water-Vapor-Proof, Flexible"
MIL-P-27401-B	"Propellant Pressurizing Agent, Nitrogen"
AFETR M-127-1	"General Range Safety Manual"

*MOSE applicable documents - utilized for specifications or as guidelines for MOSE design.

appear, additional safeguards such as "dead-man" switches are installed and interlocks to prevent MOSE operation beyond design limits. Guidelines established by the Human Engineering Design Criteria for Aerospace Systems and Equipment, MIL-STD-803A-1, will be applied.

16.3.3 Spacecraft Interfaces

Design features are incorporated which physically protect the spacecraft and its systems from MOSE failure or malfunction. Materials used in the MOSE present no hazard to the spacecraft during any operational phase. Plating and bearing design take into account potentially degrading metal matings and the various environmental conditions possible. Normal operation of the MOSE will not violate the cleanliness requirements.

Shock and vibration damping are provided to protect the spacecraft and its components during checkout, transport, and launch.

16.3.4 Cleanliness Standards

The MOSE planned for use in the clean room is designed for ease of cleaning. The equipment is designed to prevent particle contamination of the clean area by proper surface treatment, materials selection, and avoidance of irregular surfaces. The equipment is also compatible for use within a class 10, 000 clean room as specified by Federal Standard 209.

The spacecraft and components must be maintained in this clean condition at all times, including during shipment and intraplant handling. Containers and covers will be designed to maintain flight items in the required clean condition. Air conditioning and monitoring units, utilized in conjunction with spacecraft containers, are also necessary.

The spacecraft will be biologically decontaminated with ETO twice during assembly and test. After the final decontamination, cool and sterile nitrogen must be circulated about the spacecraft. The MOSE will provide the units required for the complete ETO decontamination process and for the supply and control of sterile nitrogen.

16.3.5 Test Considerations

All MOSE having electrical components and used to support the spacecraft or its parts during tests use high-quality insulation, eliminating conductive paths to the test item. An electrical grounding system is provided as required, compatible with the facility and providing adequate protection to the spacecraft.

No MOSE used in the thermal-vacuum chamber will cause any contamination or function degradation of the spacecraft by outgassing, arcing, spalling, or any other means.

16.3.6 Load Factors

Strength and rigidity requirements are considered at both design (yield) and ultimate load levels. All structures have a positive margin of safety computed in accordance with MIL-HDBK-5 procedures. Figure 16-1 defines the maximum allowable load envelope for the spacecraft during all operations involving MOSE.

Limit loads are applied to the MOSE through their structural design centers of gravity and reacted statically. The reactions are appropriate to the design condition and are applied conservatively. The design of certain structural components may be dictated by either stiffness or functional limits, but analyses will verify that strength requirements are also satisfied.

There will be no evidence of excessive deflection or permanent deformation after the MOSE has been proof-loaded. Proof-load values for testing of MOSE items are equivalent to those produced by the design limit load and hazard factors.

The limit load for MOSE is normally the working load (weight of the spacecraft or component and the associated item of MOSE) multiplied by the limit load factor. Limit load factors to be used for design are listed in Table 16-2.

MOSE is designed to withstand design loads without permanent deformation or excessive deflection. Excessive deflections are those

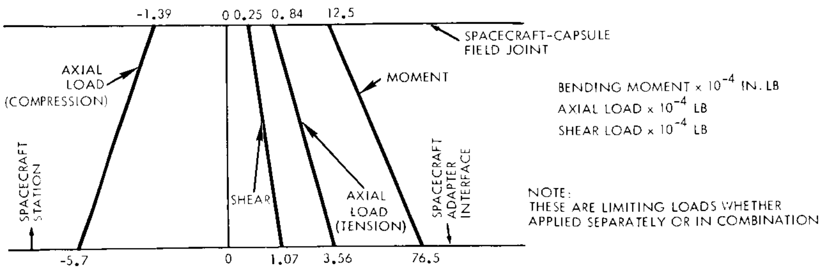


Figure 16-1

MAXIMUM ALLOWABLE FLIGHT LOAD envelope indicates loads on the spacecraft that must not be exceeded during ground handling and transportation.

Table 16-2. MOSE Limit Load Factors*

Environmental cover	N_v	+3.0	-2.0	Depending on operational sequence, wind loads may also be included
	$N_L = N_s$	±1.5	±1.5	
Shipping containers	N_v	+4.0	-3.0	
	$N_L = N_s$	±3.0	±3.0	
Handling dollies	N_v	+2.0	-1.0	
	$N_L = N_s$	±1.0	±1.0	
Assembly, handling frames, and fixtures	N_v	+4.0	-3.0	Rigidity requirements must be examined.
	$N_L = N_s$	±3.0	±3.0	
Weighing and center of gravity fixture	N_v	+2.0		All vertical forces will be assumed to vary in direction from 0 to 10 degrees from nominal rigging position.
	$N_L = N_s$	0	0	

*MOSE limit load factors - these factors are applied to meet rigidity requirements.

which would result in unsatisfactory mechanical performance or induce loads in the spacecraft or components that exceed the design loads. The design load is the limit load, multiplied by the hazard factor.

The hazard load factor for all MOSE is considered to be 1.0 except for hoisting equipment, in which case the hazard factor is to be 1.5. Pressure vessels used in MOSE will use a hazard factor of safety of 2.0.

MOSE is designed to withstand ultimate loads without failure. Failure is defined as inability to sustain ultimate load. The ultimate load is the design load multiplied by the ultimate factor of safety. The ultimate factors of safety are as follows:

Factors of Safety

Item	Hazard Factor	Ultimate Factor
All MOSE (except hoisting equipment)	1.0	2.0
Hoisting equipment (i.e., rotation or tilt fixtures, engine and propulsion handling fixtures, slings)	1.5	2.0
Pressure vessels	2.0	2.0

All MOSE that will be transported by air is designed to withstand accelerations from emergency landings without any major component breaking loose and without external physical collapse. The ultimate load factors, in accordance with MIL-A-8421B, are shown in Table 16-3.

The loads applied to the spacecraft or its components by the MOSE during movement will not exceed the flight acceptance test level spectra of shock, vibration, and acceleration.

Table 16-3. Emergency Landing Ultimate Load Factors*

Direction	Condition		
N_v	4.5	0	0
N_s	0	±8.0	0
N_L	0	0	±8.0

Note: The axes of the coordinate system are identified relative to earth and applied to mechanical handling and test equipment in their normal attitude relative to earth. The sign convention refers to direction of acceleration of the mass being handled by the equipment.

N_v = Vertical load, axis vertical relative to earth, positive action down

N_L = Lateral load, axis horizontal relative to earth and in direction of motion of the equipment

N_s = Side load, axis horizontal relative to earth and perpendicular to the direction of motion of the equipment

*Emergency landing ultimate load factors - applied to all air-transported MOSE.

17. MOSE FUNCTIONAL DESCRIPTIONS

17.1 SYSTEM FUNCTIONAL DESCRIPTION

Table 17-1 lists the system MOSE. All of the major items in the table are then described by means of summary page-long functional descriptions. Where known, part numbers and sources are given for GFE items. Thus, GFE/GAEC means that an existing item provided by Grumman Aircraft Engineering Corporation for another program can also be used for Voyager.

Table 17-1. System Level Equipment List

Number	Title	GFE or Capital Equipment	TRW**	KSC	Spare	Total***
*1	Spacecraft handling ring set		5	3		5
2	Spacecraft sling		2	2		4
*3	Spacecraft mobile stand		5	3		6
*4	Hydraset, 1 ton	CAP	2			2
*5	Hydraset, 5 ton	CAP	1			1
*6	Hydraset, 10 ton	CAP	1			1
*7	Hydraset, 20 ton	CAP	1			1
*8	Spacecraft work stand		5	3		5
9	Mechanics tool kit	CAP	30	10	10	50
10	Capsule weight and c. g. simulator	GFE	1	1		1
*11	Component alignment instruments		2	2		4
*12	Alignment optical instruments		2	2		4
*13	Spacecraft transfer fixture		1			1
14	Miscellaneous shipping container pack		10	10	5	15
*15	Vibration test fixture		1			1
16	Torque-theta test equipment		1			1
*17	Thermal-vacuum test adapter		1			1
18	Shroud-spacecraft clearance and measuring instrumentation		1	1		1
19	Decontamination shroud	GFE	1			1
20	Flight shroud planetary vehicle transporter (H14-173)	GFE/ NAA		1		1
21	Flight shroud section sling	GFE	1	1		1
22	Flight shroud planetary vehicle cover			2	1	3
*23	Flight shroud planetary vehicle hoist beam			1		1
24	Flight shroud assembly fixture			3		3
25	Instrumentation unit		1	1	1	2
*26	ETO Decontamination unit		1	1		2
27	Saturn V instrumentation unit simulator	GFE		1		1
*28	Equipment mounting panel handling fixture		12	4		16
29	Equipment mounting panel hoist sling		2	1		3
30	Equipment mounting panel installation fixture		2	1		3
*31	Air conditioning unit, transporter		1	1		2
32	Assembly air-conditioning unit		5	3		5
33	Module shipping container		2			3
34	Module shipping container sling		2			2
*35	Adapter stand		1	3		3
*36	Spacecraft horizontal stand		1			1
37	Spacecraft environmental cover		1		1	2
38	Nitrogen cooling units			3		3
39	Bogie, spacecraft mobile stand		1			1
*40	Level loading cargo lift trailer (420-63250)	GFE/GAEC	1	1		2
*41	Spacecraft shipping container		1	1		2
*42	Hoist kit (DSV-4B-303)	GFE/ DAC	1	1		1
*43	Handling rings (DSV-4B-462)	GFE/ DAC	1	1		1
*44	Roller kit (DSV-4B-1863)	GFE/ DAC	1	1		1
*45	Air carry support system (DSV-4B-1859)	GFE/ DAC	1	1		1
*46	Aircraft tie-down kit (DSV-4B-1861)	GFE/ DAC	1	1		1
*47	Aircraft access kit (DSV-4B-1860)	GFE/ DAC	1	1		1
*48	Prime mover	GFE	1	1		2
*49	Transporter cradles (DSV-4B-301)	GFE/ DAC	1	1		1
*50	Spacecraft transporter (DSV-4B-300)	GFE/ DAC	1			1
51	TRW electric prime mover		2	2		4
52	Vacuum cleaner		1	1		2
53	Ring sling		1			1

*Major MOSE items which are described in Figures 17-1 through 17-15.

**Those items utilized off-site, such as White Sands Test Facility, are listed in this column.

***Sum of items is not always equal to total quantity. Same unit may be used at more than one site.

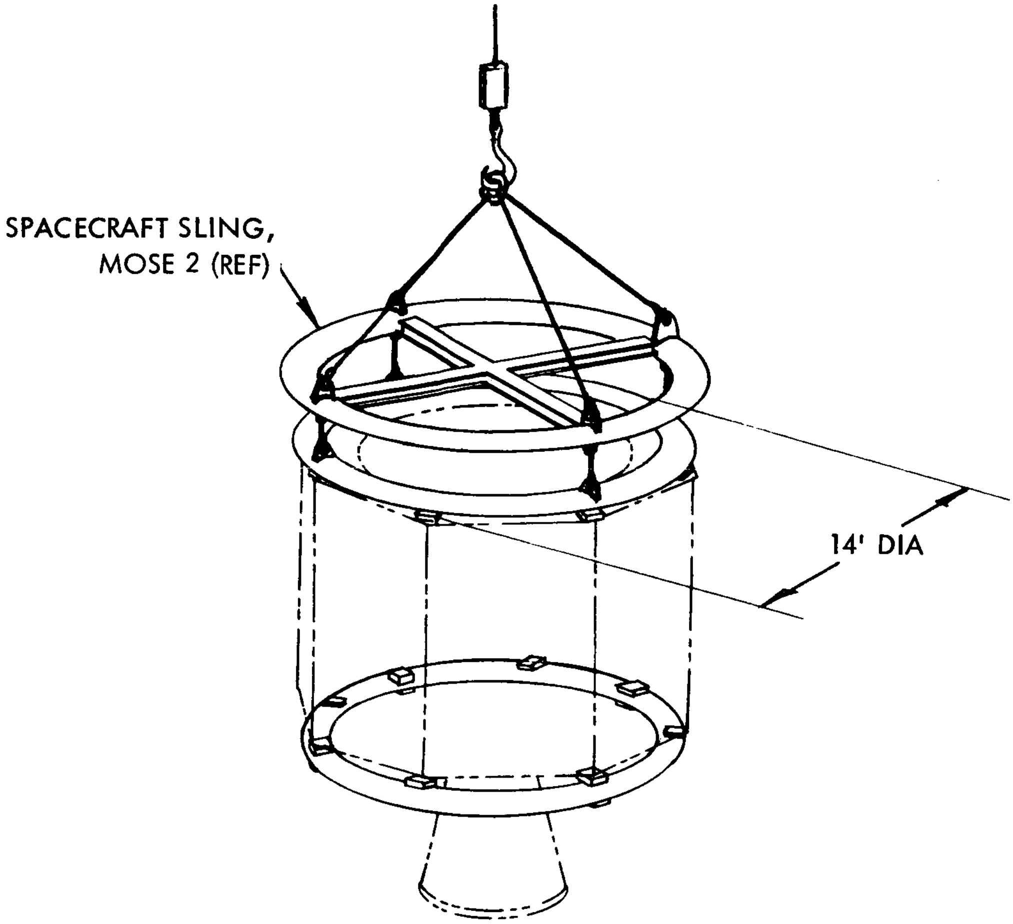


Figure 17-1. Spacecraft Handling Ring Set, MOSE 1

Functional Requirements. The handling ring set is fastened to the spacecraft during preliminary operations and remains with the spacecraft throughout most of the assembly and test flow.

Design Requirements. The rings fasten to the upper and lower surfaces at the spacecraft, and with other handling equipment, and will support the spacecraft in any orientation.

Description. Each ring is a rigid member which supports eight equally spaced brackets that match the vehicle hardpoints. The rings have sets of lifting eyes in two orientations, and mounting pads to match other support equipment.

Test Requirements. Fit, functional and proof load tests are required.

Interface Definitions. The rings interface with the spacecraft. They also interface with the spacecraft horizontal stand (MOSE 36) and with the spacecraft sling (MOSE 2).

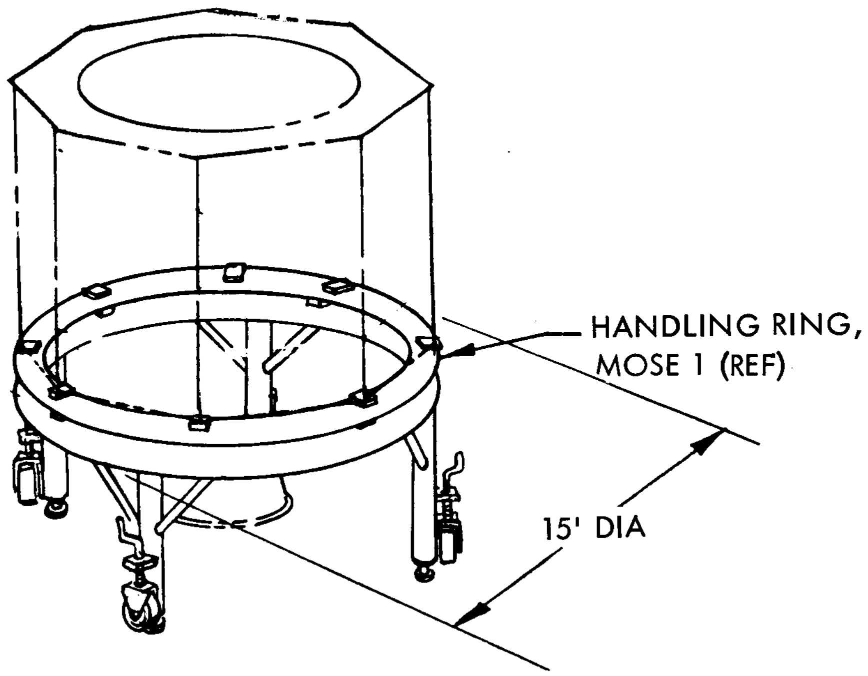


Figure 17-2. Spacecraft Mobile Stand, MOSE 3

Functional Requirements. The spacecraft mobile stand is required for support of a planetary vehicle, a spacecraft, or a propulsion module alone.

Design Requirements. The stand must support a load of 14,000 pounds. The stand may be towed, for in-plant movement. Leveling jacks are provided in the stand columns. The construction of the stand allows the antenna to be deployed without interference.

Description. The spacecraft mobile stand is a tubular ring supported by four columns. Eight pads cantilevered inward from the ring match the vehicle hardpoints. When mobility is required, casters mounted to the four columns are lowered to the floor by built-in screw jacks. A removable tow bar attaching to two of the legs is also provided.

Test Requirements. Fit checks, functional tests, and proof load tests are required on the stand.

Int' rface Definitions. The stand interfaces with the spacecraft hardpoints a..d is compatible with the TRW prime mover MOSE 52 and spacecraft work stands MOSE 8.

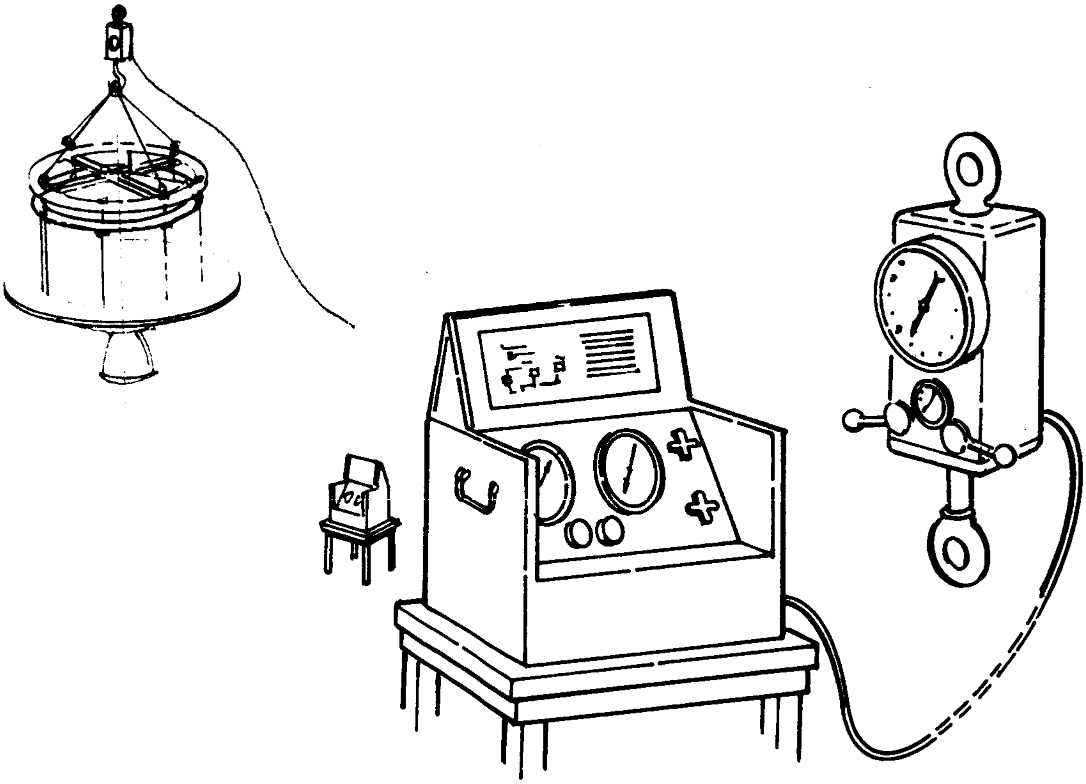


Figure 17-3. Hydrasets, MOSE 4, 5, 6, and 7

Functional Requirements. Hydrasets are required for precision control during all hoisting operations, especially those involving mating of heavy components and the movement of the spacecraft.

Design Requirements. The hydrasets must provide precision control for all hoisting operations, with both direct and remote actuation. Instruments of 1-, 5-, 10-, and 20-ton capacity must be provided.

Description. The hydraset is a hydraulic-pneumatic lifting device which provides precise control of initial and terminal hoist movement. It consists of a hydraulic cylinder with manual controls and a remote pneumatic control console.

Test Requirements. Functional testing at the rated load of each unit will be required.

Interface Description. The hydrasets interface with the facilities hoists and all handling and hoisting fixtures and slings.

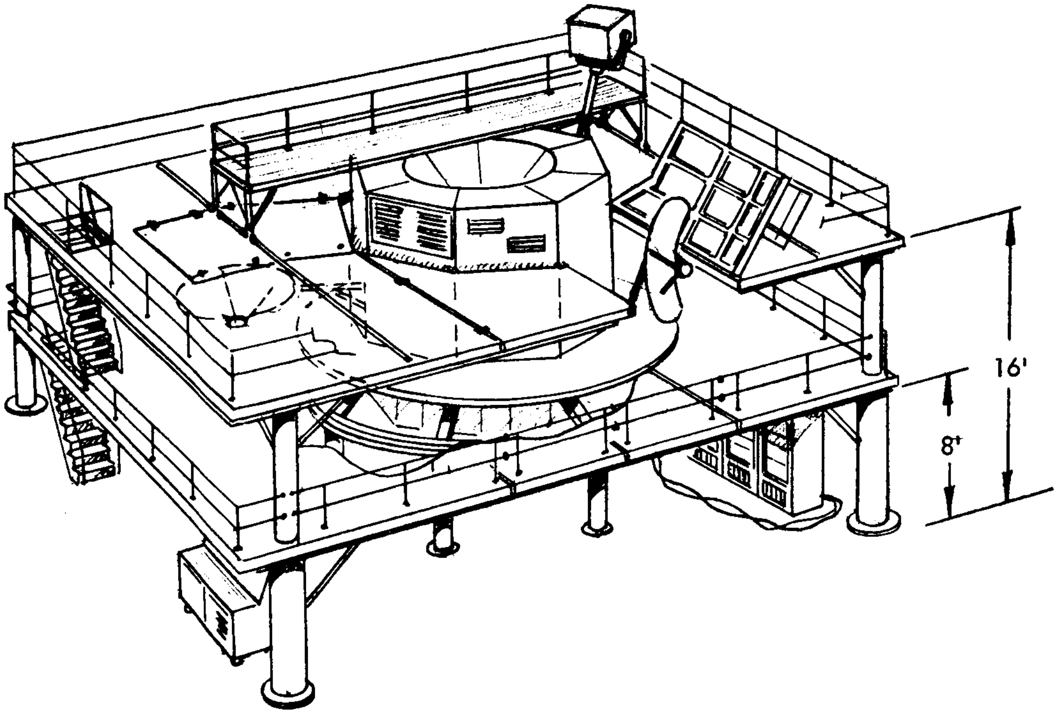


Figure 17-4. Spacecraft Work Stand, MOSE 8

Functional Requirements. A workstand is required to provide personnel access to all parts of a spacecraft, and to support all types of test and assembly equipment during assembly and checkout of the spacecraft.

Design Requirements. The workstand has working levels so as to provide complete access to the spacecraft, with ceiling heights comfortable to personnel. The workstand is configured to facilitate insertion or removal of a spacecraft. The workstand meets cleanliness requirements, and all electrical lines will be recessed into conduits giving minimal interference to personnel and equipment. The floor loading will be 1440 psf.

Description. The workstand is at two levels, with a third roll-a-way platform for access to the top of the spacecraft. Ceiling height is eight feet. The floors are steel, spanning between support columns. The number of columns, cross-beams, and truss-work is minimized to reduce obstructions. Fold away portions of the floor give ready ingress and egress of spacecraft. Safety rails are about the outer edge of both levels. Stairs provide access to levels for personnel. The workstand can be utilized as a single unit for servicing four spacecraft or a portion can be detached to service one spacecraft.

Test Requirements. Capability with facility features will be determined and fit and functional tests are required.

Interface Definitions. The workstand interfaces with the spacecraft and the spacecraft mobile stand (MOSE 3).

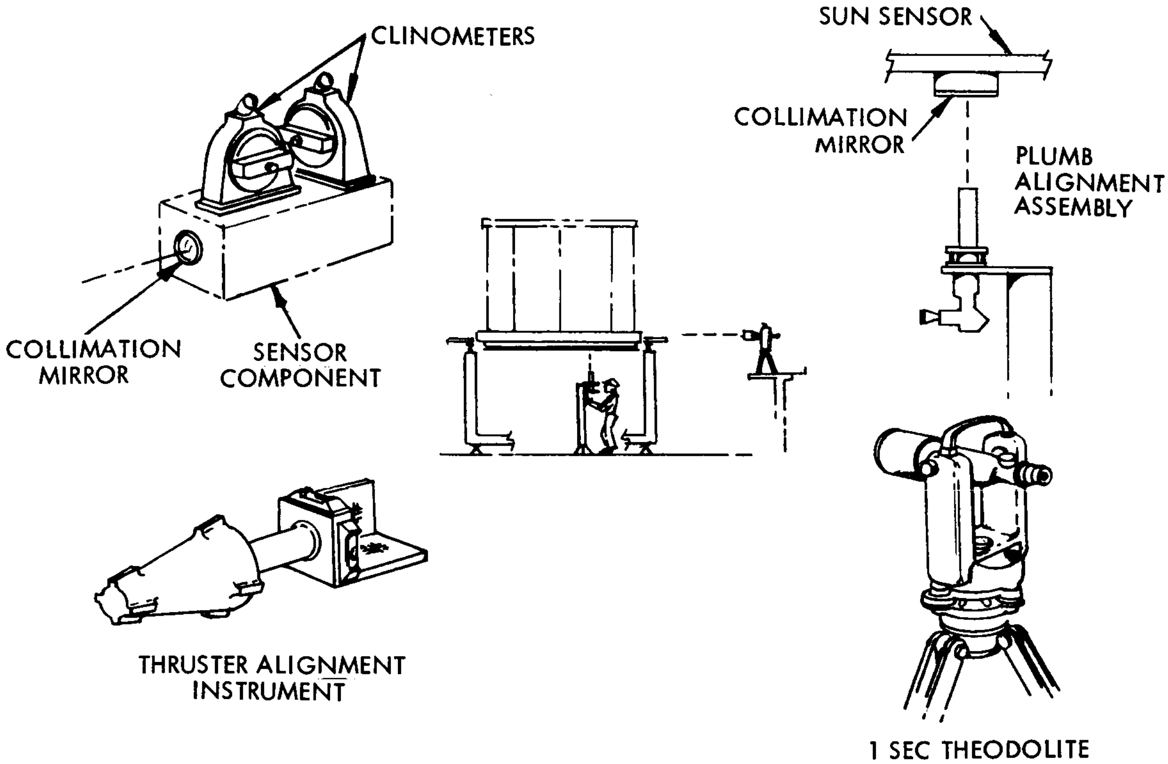


Figure 17-5. Component Alignment Instruments, MOSE 11, and Alignment Optical Instruments, MOSE 12

Functional Requirements. Many spacecraft components (e.g., the control system sensors, the propulsion system) require careful alignment to the spacecraft reference system. Optical alignment instruments and their associated fixtures for mounting to each of the components is provided for this purpose.

Design Requirements. The optical instruments are standard commercial instruments with standard fittings and with pointing accuracies of about 1 sec of arc. The component alignment instruments must attach easily and accurately without affecting the alignment of the components on which they are mounted.

Description. The optical instruments are commercially available 1-second theodolites and optical plumb alignment assemblies. The component alignment instruments are fixtures to adapt the optical instruments to a specified spacecraft component.

Test Requirements. Optical bench calibration procedures and accuracy calibration of the alignment instruments will satisfy test requirements.

Interface Definition. The alignment instruments interface with each other and the spacecraft.

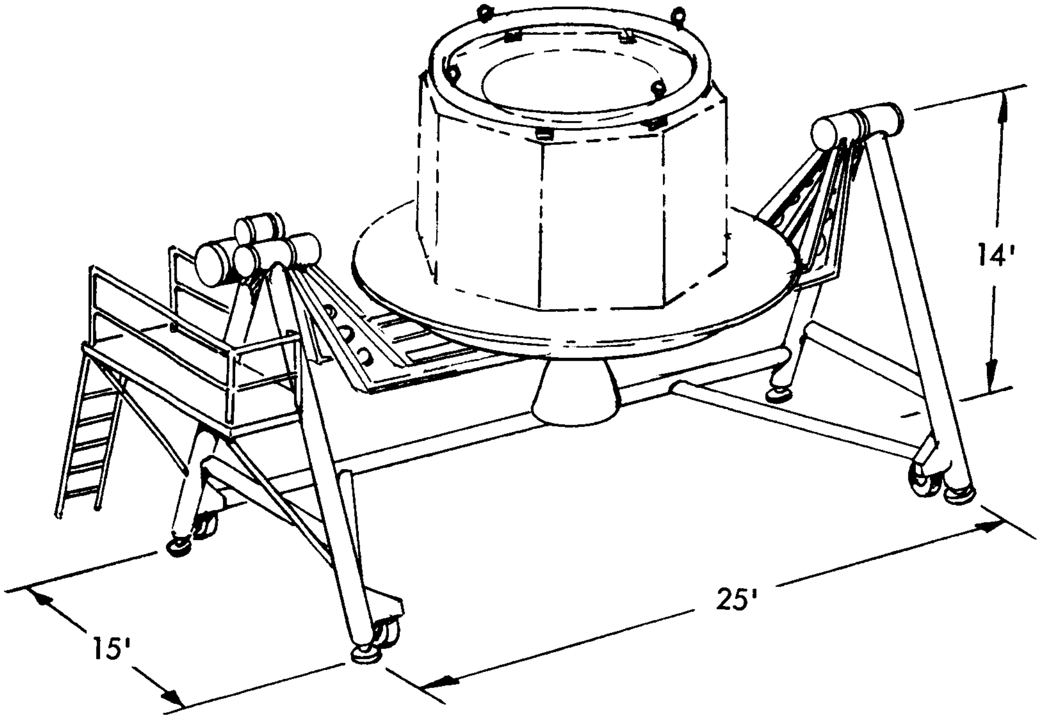


Figure 17-6. Spacecraft Transfer Fixture, MOSE 13

Functional Requirements. The spacecraft transfer fixture is required to provide for spacecraft rotation from the upright to the inverted and horizontal position.

Design Requirements. The spacecraft shall be rotated at a slow rate to assure that it is not adversely affected. Attachment points for the planetary vehicle shall be such that safe handling is inherent.

Description. The spacecraft inverter is composed of a basic A-frame which supports a spacecraft mounting ring. The ring is offset from the swivel points of the A-frame by two arms. This offset provides rotation of the spacecraft about its center of gravity in the horizontal axis. The support arms are attached to the A-frame at bearing points and are rotated bv a gear box which also serves as a locking and positioning device. A tow bar and set of retractable caster wheels at each member of the A-frame provide mobility. Four jacks level and secure the inverter.

Test Requirements. The equipment requires testing to insure that it can support the required loads and that the spacecraft will fit and rotate without interference.

Interface Definitions. The spacecraft transfer fixture interfaces with the handling ring set (MOSE 1).

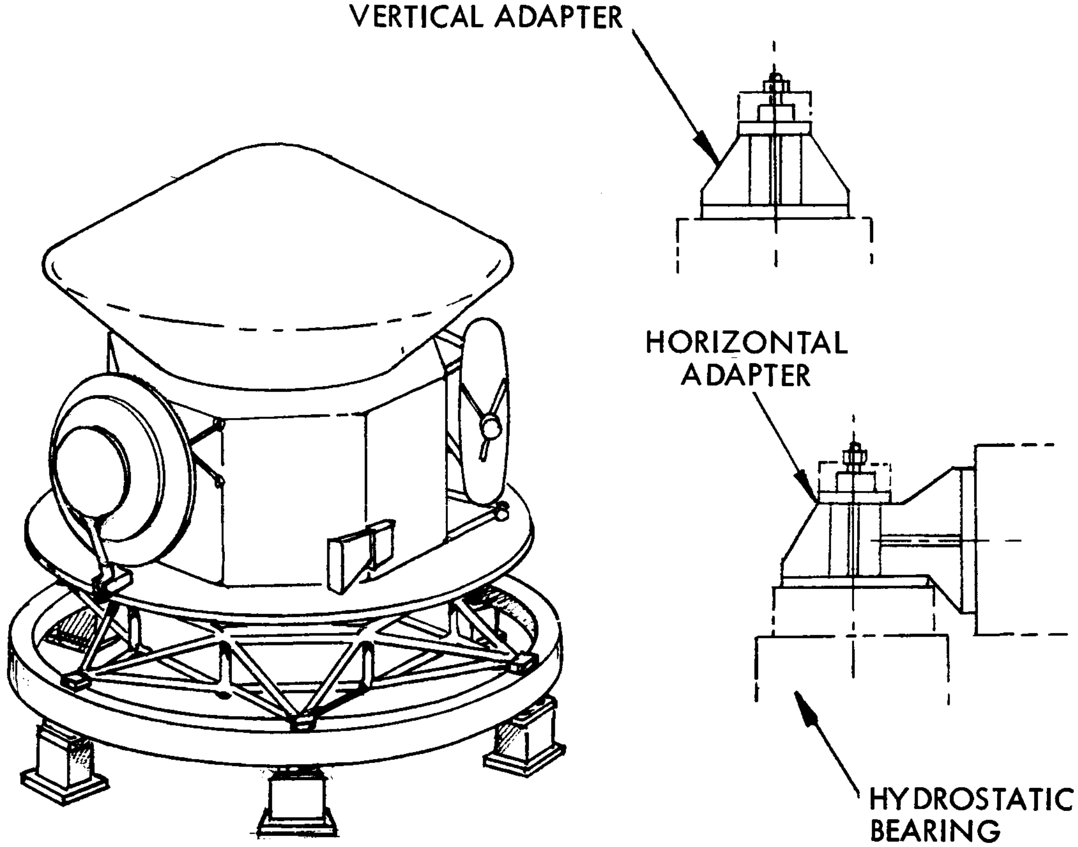


Figure 17-7. Vibration Test Fixture, MOSE 15

Functional Requirements. The vibration test fixture will be used for securing a completed spacecraft, with flight adapter and simulated capsule, to the vibration machines. The testing is in both vertical and horizontal directions.

Design Requirements. The vibration test fixture is designed to support a load of 31,000 pounds.

Description. The fixture consists of a rigid ring which interfaces with the outermost hardpoints of the flight adapter. Four removable adapters attached to the ring connect to the shaker units. One set of adapters is for vertical vibrations, another set is for horizontal vibration. The fixture rests on hydrostatic bearings during horizontal vibration tests.

Test Requirements. The fixture requires fit checks, functional tests, and proof load tests.

Interface Definitions. The fixture ring interfaces with the flight adapter. The horizontal adapter set interfaces with hydrostatic bearings and with the shaker units.

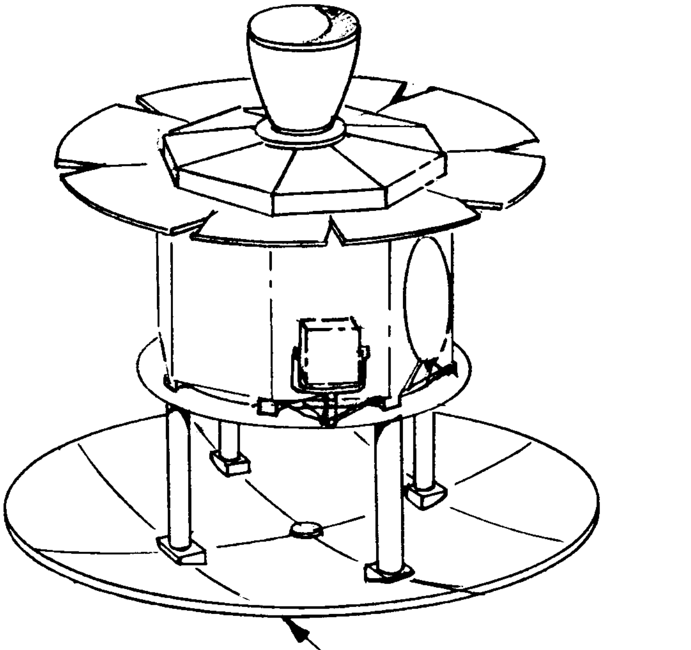


Figure 17-8. Thermal-Vacuum Test Adapter, MOSE 17

Functional Requirements. The thermal-vacuum test adapters are required for supporting the spacecraft in an inverted position during thermal-vacuum tests.

Design Requirements. All material must be selected for low temperature, outgassing, radiation, and conduction characteristics. Shields and/or paint must be utilized to reduce radiation effects. Heaters must be used to equalize the effect of conduction between the planetary vehicle and the thermal-vacuum test adapters.

Description. The adapters support the spacecraft from the thermal-vacuum cover (vertical) and will be attached to the chamber cover with mounting pads. Insulated mounting pads or adapters between the basic structure and the spacecraft reduce heat conduction. Heater elements are used to equalize thermal conditions.

Test Requirements. Tests must be performed on the thermal-vacuum test adapters to demonstrate design load and structural requirements, thermal conductivity and radiation effects, and compatibility requirements.

Interface Definitions. The thermal-vacuum test adapters are compatible with the spacecraft and the thermal-vacuum chamber and cover.

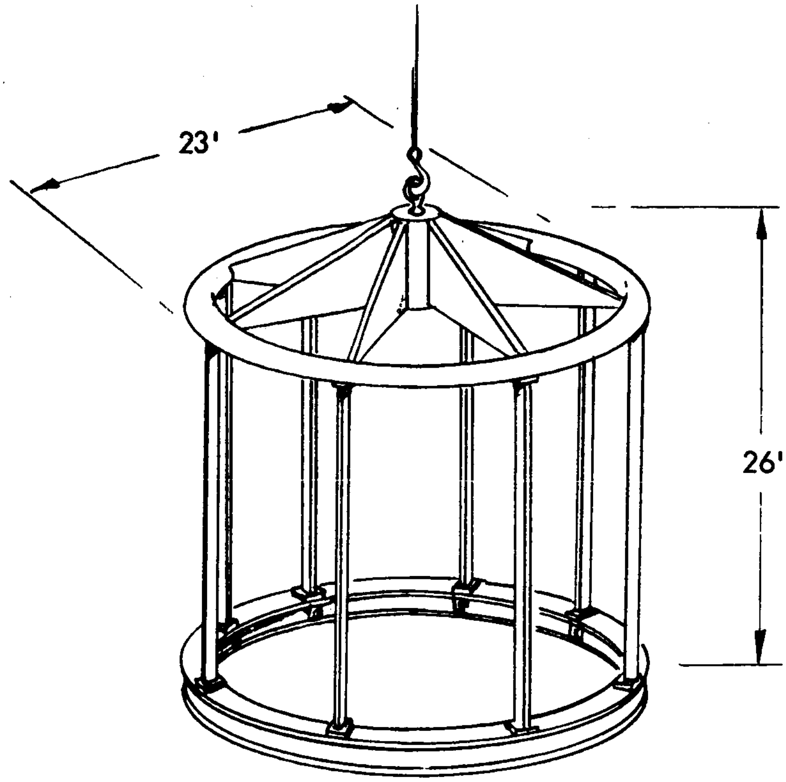


Figure 17-9. Flight Shroud/Planetary Vehicle High Beam, MOSE 23

Functional Requirements. A hoist beam is required to lift an unfueled planetary vehicle encapsulated within a flight shroud.

Design Requirements. The hoist beam is configured to utilize the available hardpoints about the lower portion of the periphery of the flight shroud. The hoist beam is capable of lifting 20,000 pounds.

Description. The hoist beam consists of two horizontal rings one above the other, the two attached together by eight equally spaced stringers. The lower ring has lugs to connect to the shroud hardpoints. The upper ring is stiffened by eight cantilever beams radiating from a center post. A lifting eye is fixed to the center post.

Test Requirements. The hoist beam requires fit, functional, and proof-load tests.

Interface Definitions. The hoist beam interfaces with the shroud field joint mating holes. The beam is also compatible with use of the shroud-planetary vehicle cover (MOSE 18).

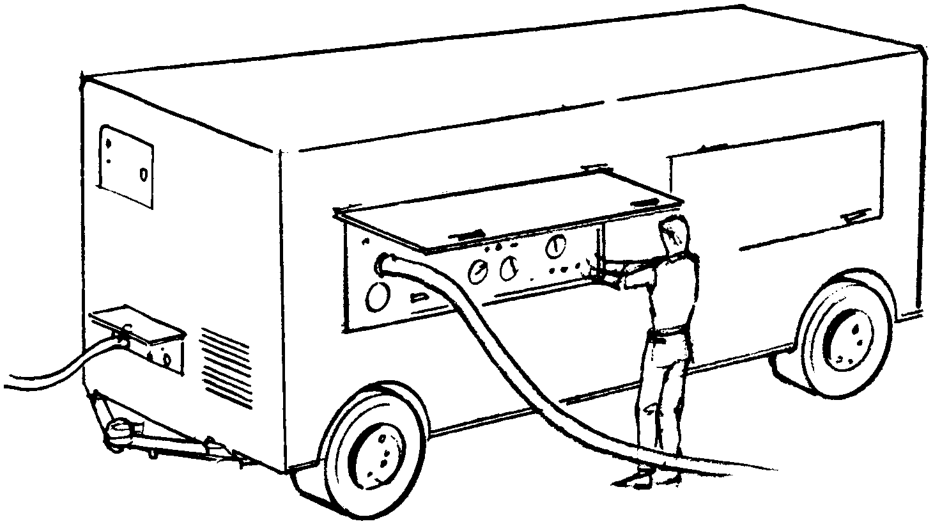


Figure 17-10. ETO Decontamination Unit, MOSE 26

Functional Requirements. The ETO decontamination unit is necessary to assure control of the biological and particulate contamination level of the spacecraft, during assembly, test and pre-launch operations.

Design Requirements. The unit functional requirements are as follows:

- To supply dry nitrogen gas to the encapsulated planetary vehicle at a temperature of 90 ±5°F and a pressure of 0.5 psig. This gas is required to preheat the planetary vehicle assembly to a stabilized temperature of 86°F prior to the initiation of the ETO decontamination cycle.

- To supply ETO gas (12 percent ethylene oxide, 88 percent Freon 12 by weight) to the encapsulated planetary vehicle at a temperature of 86 to 95°F, a pressure of 0.5 psig, and a relative humidity of 40 ±5 percent.

- To recirculate and replenish ETO gas through the encapsulated planetary vehicle for a period of 11 to 18 hours, maintaining the prescribed temperature, pressure, and relative humidity. The length of the ETO cycle will be determined from assay of the biological load of the planetary vehicle.

- To dispose of excess ETO gas and all remaining gas within the planetary vehicle compartment after completion of the ETO cycle.

- To supply and circulate sterilized dry nitrogen purge gas at a temperature of 90 ±5°F in order to purge the encapsulated planetary vehicle after the ETO decontamination has been completed.

Description. The unit consists of a remote control and monitoring console, a gas control and conditioning unit, and quick-acting connections to the shroud-spacecraft assembly and sources of LN_2 and Freon-ethylene oxide (88-12).

Test Requirements. The unit will be leak and pressure tested and functionally tested.

Interface Definition. The unit has pneumatic and instrumentation interfaces with the shroud-spacecraft assembly.

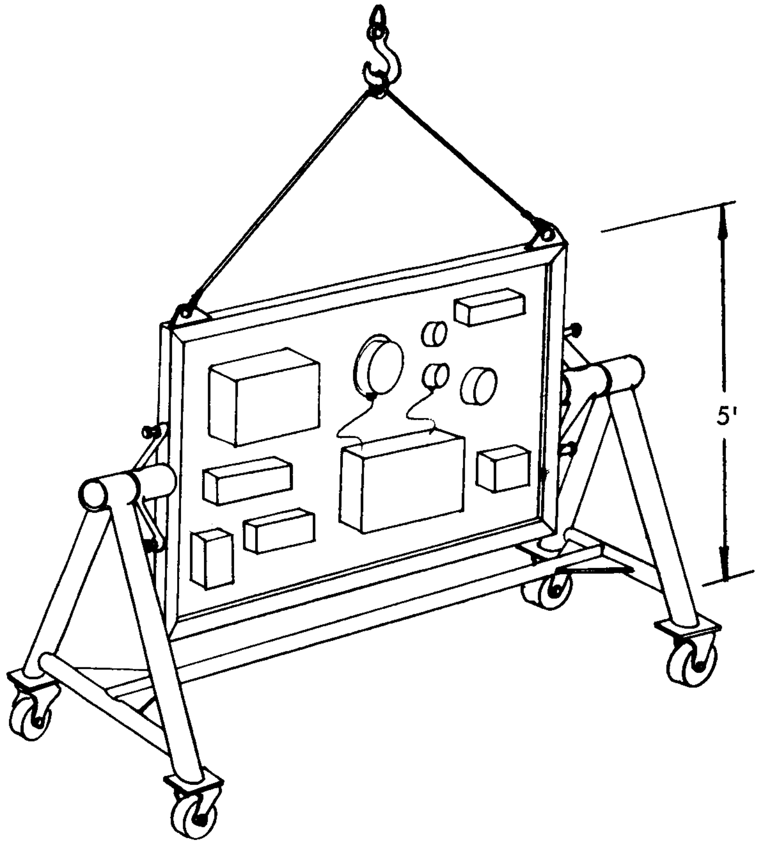


Figure 17-11. Equipment Mounting Panel Handling Fixture, MOSE 28

Functional Requirements. A fixture is required to support and handle the equipment mounting panels and to provide in-plant mobility.

Design Requirements. The handling fixture must support individual equipment panels and adjust to panels of varying outside dimensions. It must rotate the panels 360 degrees and hold them at any angle during rotation.

Description. The fixture consists of a tubular structure mounted on four rubber-tired wheels. The structure carries bronze bearings on which a panel holding frame rotates. This frame surrounds the panel and supports it at its outer edges. Eyebolts are provided for handling the panels by means of a hoist sling. Bolt clearance holes are provided for transfer of panels to the equipment mounting panel installation fixture (MOSE 30).

Test Requirements. The fixture requires fit checks and functional tests.

Interface Definitions. The fixture is compatible with the equipment mounting panels. It interfaces with the equipment mounting panel hoist sling (MOSE 29), and is used in conjunction with the equipment mounting panel installation fixture (MOSE 30).

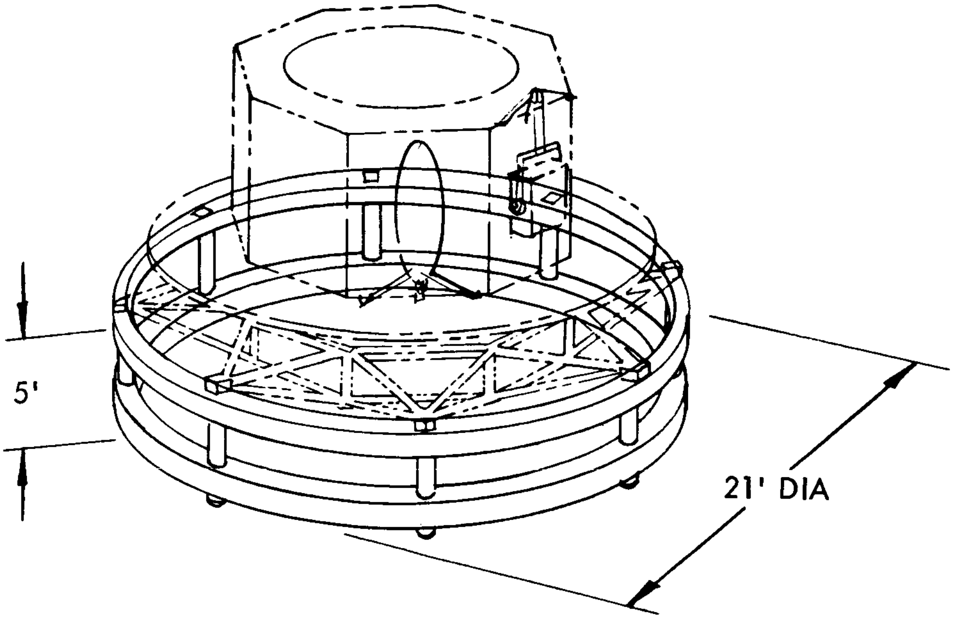


Figure 17-12. Adapter Stand, MOSE 35

Functional Description. The adapter stand interfaces with the flight adapter and is capable of supporting the capsule, spacecraft, adapter and shroud stack.

Design Requirements. The stand will support a weight of 20, 000 pounds.

Description. The stand consists of two rings the diameter of the flight adapter. Vertical columns separating the rings are spaced according to the outermost adapter support points. The column heights are adjustable to provide leveling of the stack.

Test Requirements. Fit, functional, and proof load tests are required.

Interface Definitions. The stand interfaces with the flight adapter.

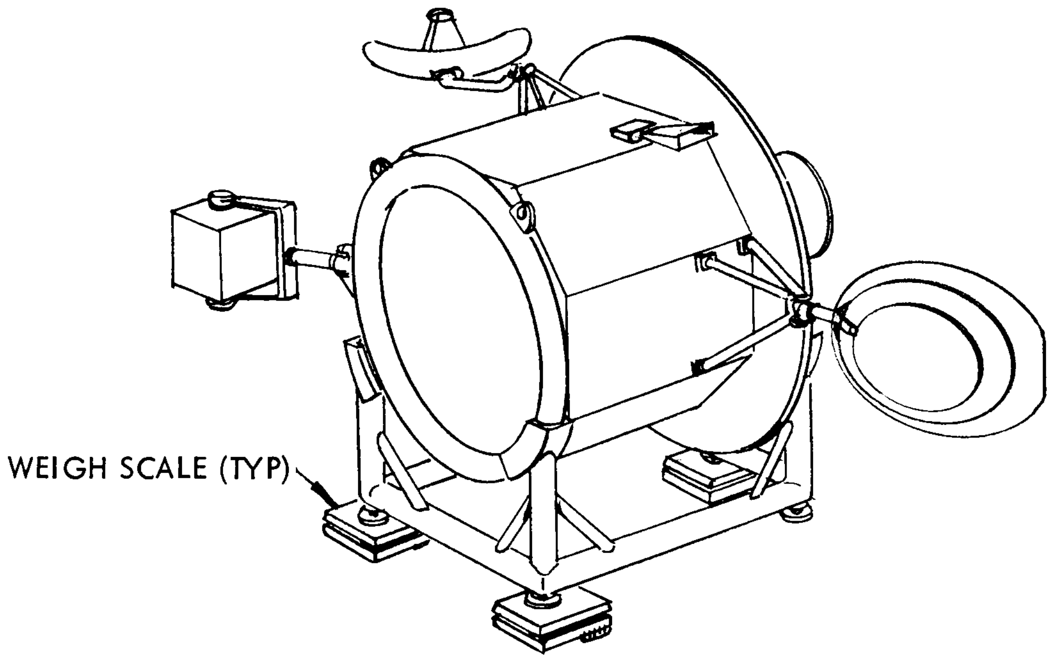


Figure 17-13. Spacecraft Horizontal Stand, MOSE 36

Functional Description. The horizontal stand supports the spacecraft on its side during weight and center of gravity testing and during appendage deployment testing.

Design Requirements. The stand will support a spacecraft on its side. The stand will support a load of 6000 pounds.

Description. The stand is a rectangular base frame with four vertical columns extending up. The upper ends of the columns are bracketed to match the pads of the spacecraft handling ring set. The base of each column contains a jack pad to provide alignment of the vehicle. Retractable, shock absorbing casters are fixed to each column, providing mobility to the stand.

Test Requirements. Fit, functional and proof load tests are required.

Interface Definitions. The stand interfaces with the spacecraft handling ring set (MOSE 1), and the TRW electric prime mover (MOSE 52).

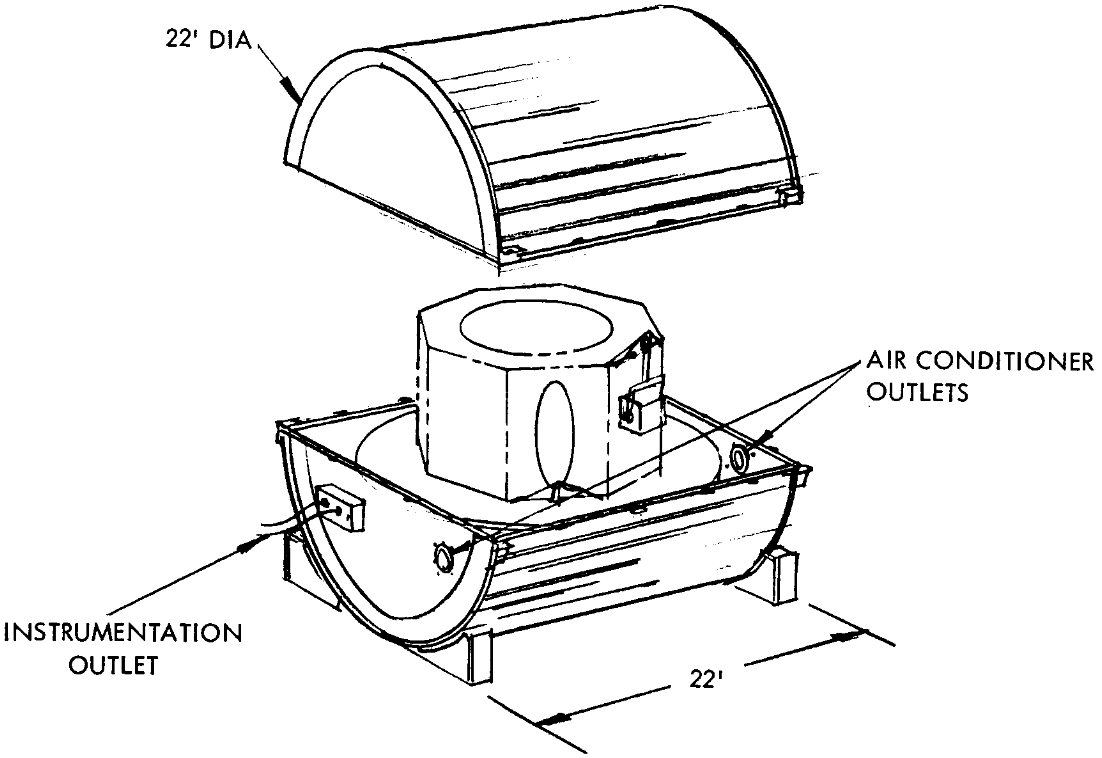


Figure 17-14. Spacecraft Shipping Container, MOSE 41

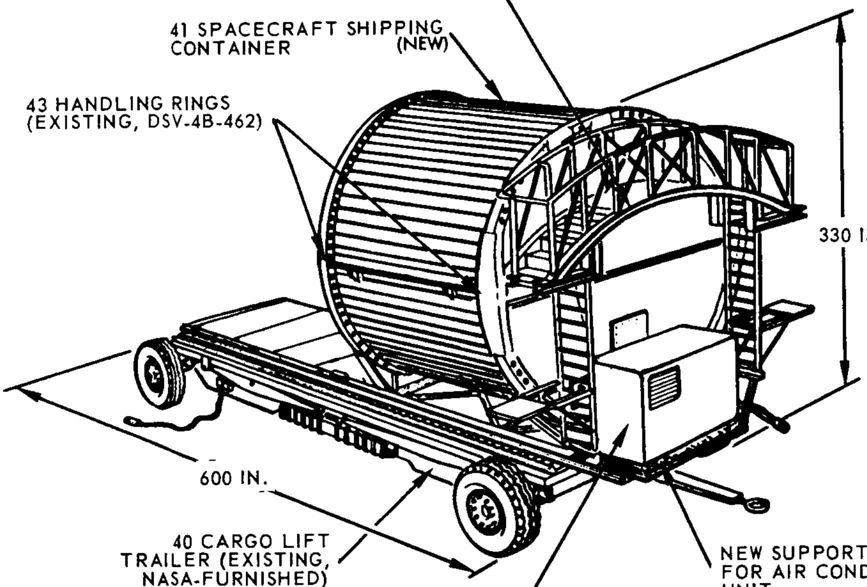
Functional Requirements. The spacecraft, during storage and shipment must be protected from acceleration loads and environmental hazards.

Design Requirements. The shipping container protects the spacecraft during transportation and storage and mounts it in its flight attitude. It provides environmental protection to meet temperature, humidity, and cleanliness-controlled environments in accordance with FED-STD-209 for class 10,000 cleanliness.

Description. The spacecraft shipping container consists of two sections which together form a cylindrical container. The spacecraft mounting points are an integral part of the lower portion of the container. The mounting points are shock absorbing. The two sections are closed at their ends and joined by a series of latches forming an airtight and watertight seam.

Test Requirements. The shipping container will be proof-load tested to demonstrate its design and fabrication adequacy, and will be functionally tested to demonstrate its required performance in supporting the spacecraft and maintaining required internal environment.

Interface Definition. The spacecraft shipping container interfaces with the eight hardpoints on the spacecraft utilized to mate to the adapter. The container interfaces with the inlet and outlet duct of the air-conditioning unit (MOSE 31), and with the Super Guppy transporter group.



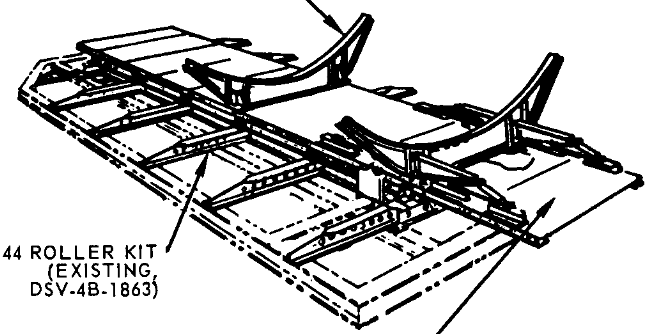


Figure 17-15. Transporter Group, Super Guppy, MOSE 25, 31, 41, 42, 43, 44, 45, 46, 47, 48, 49, 50

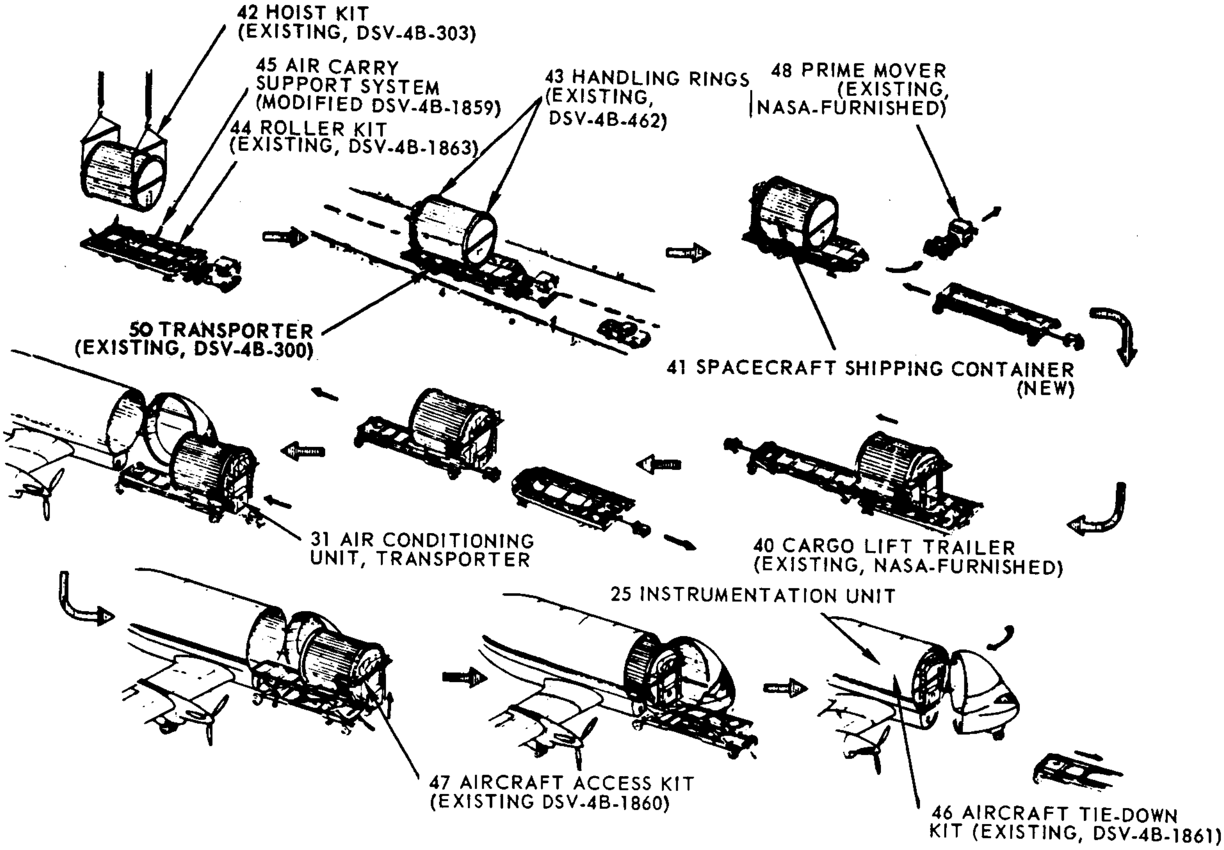


Figure 17-15. Transporter Group, Super Guppy, MOSE 25, 31, 41, 42, 43, 44, 45, 46, 47, 48, 49, 50 (Continued)

Functional Requirements. The spacecraft will be transported by suitable cargo aircraft to cross-country destinations. The spacecraft will be completely assembled (dry) and mounted and sealed within a shipping container while being transported. Stabilized support, handling, and transfer loading equipment must be provided for loading and unloading the aircraft with the spacecraft shipping container. The spacecraft must be environmentally controlled while being transferred into, secured within, and transported by the aircraft. All loads imposed upon the spacecraft during these operations must be monitored and recorded.

Design Requirements. The handling equipment and transfer procedures must not impose excess shock and vibration loads or resonant frequencies on the spacecraft. The transported group functions at altitudes from sea level to 30,000 feet. A temperature, humidity, and cleanliness-controlled environment in accordance with FED-STD-209, class 10,000 is provided within the spacecraft shipping container. The air conditioning equipment is mounted so that it remains connected to the shipping container during all handling operations. An instrumentation unit monitors and records the load history imposed on the spacecraft.

Description. The Super Guppy transporter group consists of a Super Guppy aircraft; a cargo-lift trailer (MOSE 40); a roller kit (MOSE 44); an air-carry support system (MOSE 45); an aircraft access kit (MOSE 47); an aircraft tie-down kit (MOSE 46); handling rings (MOSE 43); a hoist kit (MOSE 42); an air conditioning unit (MOSE 31); an umbilical assembly; and a spacecraft shipping container (MOSE 41). The size and weight of the Voyager spacecraft allows use of most of the Saturn S-IVB Stage SG transporting equipment with minor modifications.

The cargo-lift trailer (MOSE 40) is government-furnished equipment and requires no modification. The aircraft access kit (MOSE 47, Saturn model DSV-4B-1860) and the aircraft tie-down kit (MOSE 46, Saturn model DSV-4B-1861) require no modification. The Saturn S-IVB stage handling rings (MOSE 43, Saturn model DSV-4B-462) and the Saturn S-IVB stage hoist kit (MOSE 42, Saturn model DSV-4B-303) require no modification.

The Saturn S-IVB roller kit (MOSE 44, Saturn model DSV-4B-1863) allows for transfer of the spacecraft shipping container from the Saturn S-IVB transporter (MOSE 50, Saturn model DSV-4B-300) to the cargo lift trailer. The roller kit is modified to accommodate new cradle assembly locations and an air-conditioning unit for the spacecraft shipping container.

The air-carry support system (MOSE 45, Saturn model DSV-4B-1859) supports the spacecraft shipping container on the road transporter, the cargo-lift trailer, and in the Super Guppy. The air-carry support system is modified to accommodate new cradle locations and the air conditioning unit for the spacecraft shipping container.

The instrumentation unit (MOSE 25) consists of an acceleration switch alarm; a multiple-channel strain-gage type signal conditioner; humidity and temperature indicators; a tape recorder, and a patch panel for varying the channels on the strain-gage signal conditioner.

Test Requirements. The transporter group is functionally and proof-load tested, and the environmental conditioning equipment is qualified to assure its performance within the ambient conditions it experiences during air transportation.

Interface Requirements. The Super Guppy transportation group interfaces with the spacecraft shipping container, the Super Guppy loading rails, and the land transporter equipment group.

17.2 SUBSYSTEM FUNCTIONAL DESCRIPTION

Table 17-2 lists the subsystem MOSE. As in the system level functional descriptions, only major subsystem MOSE are included in the subsystem functional descriptions. A major number of existing items of MOSE in support of LM (identified as GFE/GAEC in the table) are specified for Voyager. The subsystem end items are numbered beginning with 101.

Table 17-2. Subsystem Level Equipment List

Number	Title	GFE or Capital Equipment	TRW**	KSC	Spare	Total***
101	S-band subsystem electronic shipping container		3	3		3
*102	High-gain antenna container		3	3		3
103	High-gain antenna sling		1	1		2
*104	Medium-gain antenna container		3	3		3
105	Medium-gain antenna sling		1	1		2
*106	Low-gain antenna container		3	3		3
*107	Relay link antenna shipping container		3	3		3
*108	Command subsystem shipping container		3	3		3
*109	Computer and sequencer system shipping container		3	3		3
*110	Telemetry subsystem shipping container		3	3		3
*111	Data storage subsystem shipping container		3	3		3
*112	Guidance and control subsystem shipping container		3	3		3
113	Guidance and control subsystem shipping container sling		1	1		2
114	Reaction control pressure vessel handling fixture		4	2		4
115	Reaction control pressure vessel handling sling		2	1		2
116	Pneumatic leak check console		2	2		2
117	Pneumatics cart		5	3		5
*118	Power subsystem shipping container		3	3		3
*119	Solar array mounting fixture		4	1		4
*120	Solar array handling dolly		4	1		4
121	Solar array hoisting sling		4	1		4
*122	Solar array protective cover		24	24	6	30
123	Dummy solar arrays		8			8
124	Solar array shipping containers		3	3		3
125	Solar array handling frame		4	1		4
126	Equipment module protective cover sling		1			1
127	Equipment module protective cover		2			2
*128	Equipment module mobile stand		3	1		3
129	Equipment module work stand		3	1		4
130	Louvers shipping container		3	3		3
*131	Louver installation and handling device					
*132	Louver protective cover		27	27	5	32
133	Insulation shipping container		3	3		3
134	Louver sling		3	1		3
*135	Ordnance checkout kit and handling case		1	3		4
136	Pyrotechnic subsystem shipping container		2	2		2
137	Pyrotechnic subsystem shipping container sling		1	1		2
138	PSP shipping container		4	4		4
139	PSP assembly and handling fixture		1	1		2
140	Fixed science package sling		1	1		2
141	Propulsion module test facility adapter		1			1
142	Portable clean environment kit (420-13130)	GFE/GAEC	1	1		2
143	Engine installation dolly (420-63400)	GFE/GAEC	1	1		2
144	Propellant system checkout unit (430-62170)	GFE/GAEC	1	1		2
145	Propulsion system checkout cart (430-62180)	GFE/GAEC	1	1		2
146	Halogen leak detector (430-62350)	GFE/GAEC	1	1		2
147	Helium-hydrogen mass spectrometer leak detector (430-82720)	GFE/GAEC	1	1		2
148	Fuel loading control assembly (430-64430)	GFE/GAEC	1	1		2
149	Oxidizer loading control assembly (430-64450)	GFE/GAEC	1	1		2
150	Pressure maintenance unit (430-64500)	GFE/GAEC	1	1		2
151	Helium transfer and container unit (430-94009)	GFE/GAEC	1	1		2
152	Helium booster cart (430-94022	GFE/GAEC	1	1		2
153	Fuel ready storage unit (430-94058)	GFE/GAEC	1	1		2
154	Oxidizer ready storage unit (430-94059)	GFE/GAEC	1	1		2
155	Fuel vapor disposal unit (430-94060)	GFE/GAEC	1	1		2
156	Oxidizer vapor disposal unit	GFE/GAEC	1	1		2
157	Helium storage trailer (430-94062)	GFE/GAEC	1	1		2
158	Propellant module protective cover	GFE/GAEC	3	3		3
159	Cover, protective engine LWD-420-63167	GFE/GAEC	3	3	1	4
160	Cover, engine skirt LWD-420-63139	GFE/GAEC	3	3	1	4
161	Dolly, engine skirt LWD-420-63400	GFE/GAEC	3	1		3
162	Plug, engine LWD-420-63420	GFE/GAEC	1	1		2
163	Support stand, engine LWD-420-1050	GFE/GAEC	4	1		5
164	Engine shipping containe-	GFE	2	1		2
165	Engine trunnion sling	GFE	2	1		2
166	Engine handling sling	GFE	2	1		2
167	Propellant module handling sling adapter	GFE	1	1		1
168	Engine work platforms	GFE	2			2
169	Engine simulator		3			3
170	C-1 engine simulator		12			12
171	Propellant tank dolly		4			4
172	Propellant tank sling		2			2
173	Helium tank assembly fixture		2			2
174	Helium tank sling		2			2
175	Battery handling fixture		2			2
176	Battery sling		1			1

*Major MOSE items which are described in Figures 17-16 through 17-25.

**Those items utilized off-site, such as White Sands Test Facility, are listed in this column.

***Sum of items is not always equal to total quantity. Same unit may be used at more than one site.

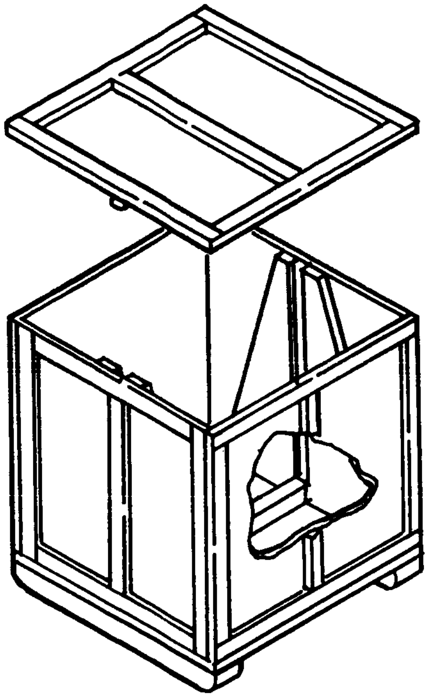


Figure 17-16. Subsystem Shipping Containers, MOSE 101, 108, 109, 110, 111, 112, and 118

Functional Requirements. The shipping containers must protect the subsystem during general handling, shipment, and storage.

Design Requirements. The containers must protect the subsystem from physical damage and environmental contamination during surface and air transportation and storage. Their weights and sizes must be the minimum necessary to provide the desired protection. Desiccants conforming to MIL-D-3716 must be used for humidity control. Breathing provisions must be incorporated for air transportation with pressure differentials from sea level to 30,000 feet, but must not compromise the cleanliness of the subsystem. The containers must be capable of being transported by rail, truck, or air, and must be reusable.

Description. The shipping containers consist of environmental covers (barrier material), shock mitigating systems, and external protective containers. The subsystems will be completely encapsulated and supported in such a manner that the load is distributed equally.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded containers as well as flat drop, rotational drop, and pendulum impact tests.

Interface Definitions. The shipping containers interface with the subsystems, but have no interface with other OSE.

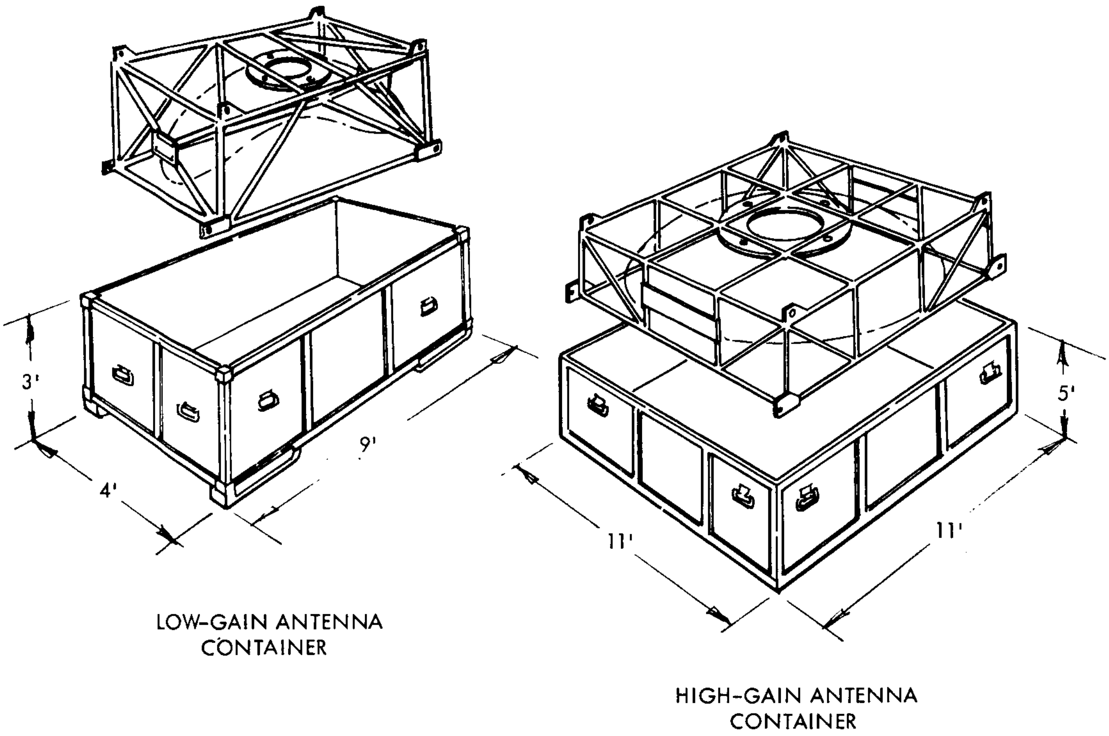


Figure 17-17. High Gain Antenna Shipping Container, MOSE 102 and Medium Gain Antenna Shipping Container, MOSE 104

Functional Requirements. The high and medium gain antenna shipping containers must protect the antennas during handling, shipping, and storage.

Design Requirements. The shipping containers must protect the antennas from physical damage and particulate contamination during surface and air transportation and storage. Their weight and size must be the minimum necessary to provide the desired protection. Shock and vibration isolation must be provided. Desiccants conforming to MIL-D-3716 must be used for humidity control. Breathing provisions must be incorporated for air transportation to equalize pressure differentials from sea level to 30,000 feet, but must not compromise the cleanliness of the antennas. The shipping containers must be reusable.

Description. Each shipping container consists of a shock-mitigating system and an exterior metal modular protective container conforming to MIL-C-22443. The antenna is hard mounted to a fixture, which in turn is shock mounted to the container on elastomer mounts.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and pendulum impact tests.

Interface Definition. The shipping container is used to store and transport the antenna but has no physical or electrical interface with other OSE.

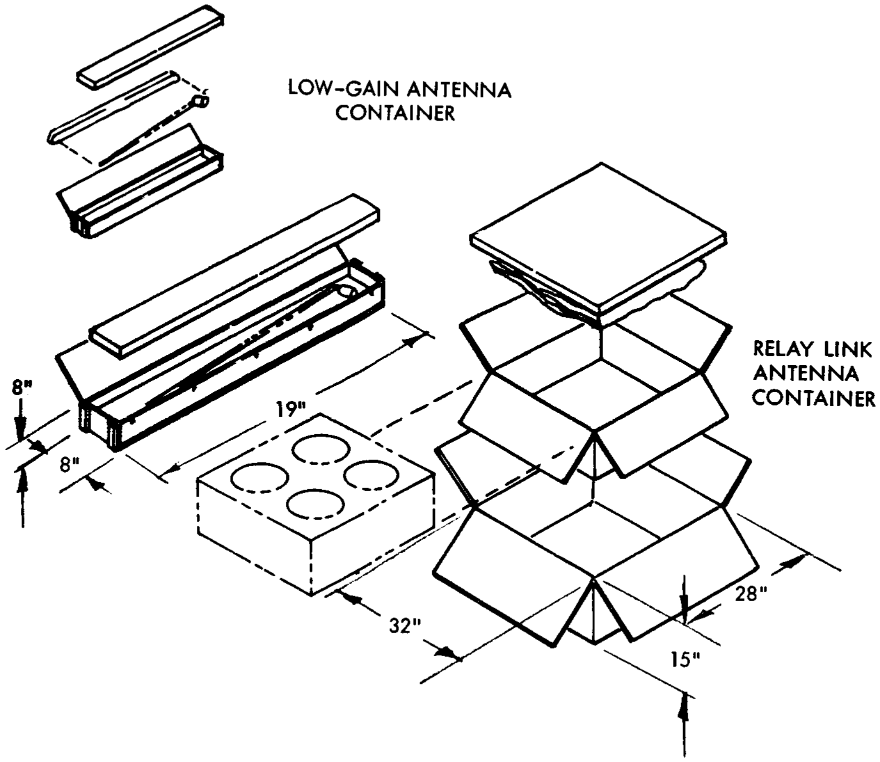


Figure 17-18. Low Gain Antenna Shipping Container, MOSE 106 and Relay Link Antenna Shipping Container, MOSE 107

Functional Requirements. The low gain and relay link antenna shipping containers must protect the antennas during handling, shipment, and storage.

Design Requirements. The shipping containers must protect the antennas from physical damage and environmental contamination during surface and air transportation and storage. Their weight and size must be the minimum necessary to provide the desired protection. Shock and vibration isolation must be provided. Desiccants conforming to MIL-D-3716 must be used to control humidity. Breathing provisions must be incorporated for air transportation to equalize pressure differentials from sea level to 30,000 feet without compromising the cleanliness of the antenna. The shipping container must be reusable.

Description. Each shipping container consists of a shock-mitigating system, an environmental cover (barrier material) and an exterior shipping container. The bagged antenna and boom are nested in a manner which distributes the load evenly.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and inclined impact tests.

Interface Requirements. The shipping container is used to store and transport the low-gain antenna but has no physical or electrical interface with other OSE.

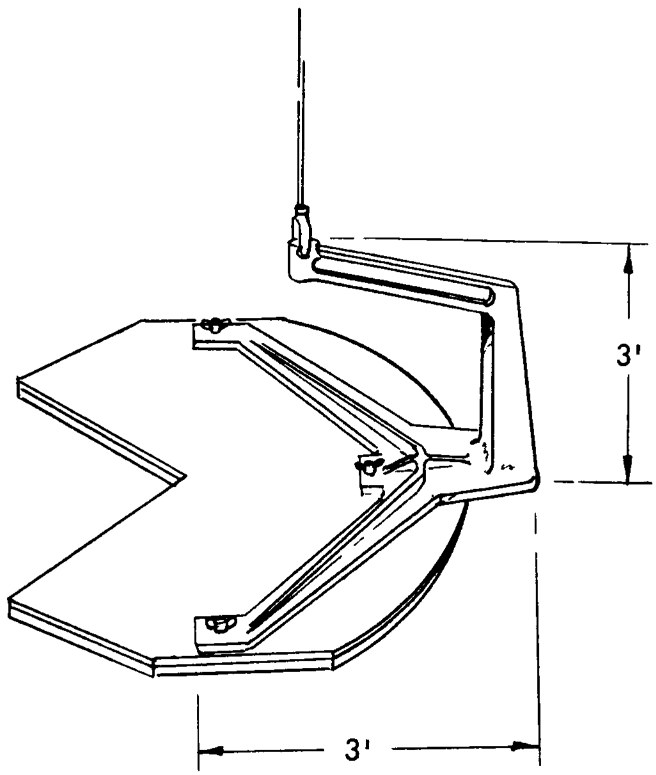


Figure 17-19. Solar Array Mounting Fixture, MOSE 119

Functional Requirements. The function of the solar array mounting fixture is to attach the array segments to the hoist for transfer from the handling frame (MOSE 25) to the spacecraft.

Design Requirements. The mounting fixture must attach to the solar array protective covers with quick-release devices and be capable of supporting the arrays in position for attachment to the spacecraft structure. It must be provided with a swivel for attachment to the hydraset (MOSE 4).

Description. The mounting fixture is a hoisting adapter of aluminum, offset to allow installation of the array segments from above.

Test Requirements. The mounting fixture will be load tested and functionally tested with dummy arrays.

Interface Definition. The mounting fixture interfaces with the solar array protective covers (MOSE 122) and with the hydraset (MOSE 4).

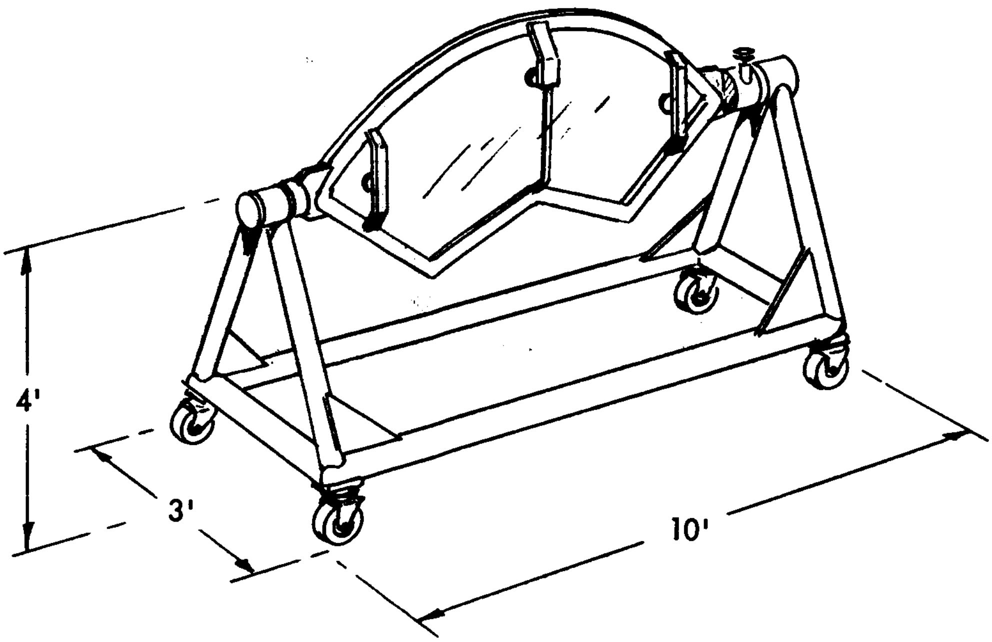


Figure 17-20. Solar Array Handling Dolly, MOSE 120

Functional Requirements. A solar array dolly is required for in-plant transport, positioning, and rotation of the solar array segments.

Design Requirements. The dolly must provide unrestricted access to both the top and bottom of the solar array panel. It must be capable of rotating the solar array 360 degrees about its transverse axis and of locking in any position. Casters must have foot-operated parking brakes and be on a broad base to provide stability. The dolly must allow the installation of the solar array protective cover (MOSE 122) and provide clearance for use of the solar array mounting fixture (MOSE 119).

Description. The solar array handling dolly is a lightweight, open frame structure of aluminum, capable of positioning the solar array for assembly, test, and integration operations, and compatible with all associated MOSE.

Test Requirements. Each dolly will be functionally tested and load tested with an assembly handling frame (MOSE 25).

Interface Definition. The solar array handling dolly has a mechanical interface with the solar array handling frame (MOSE 25).

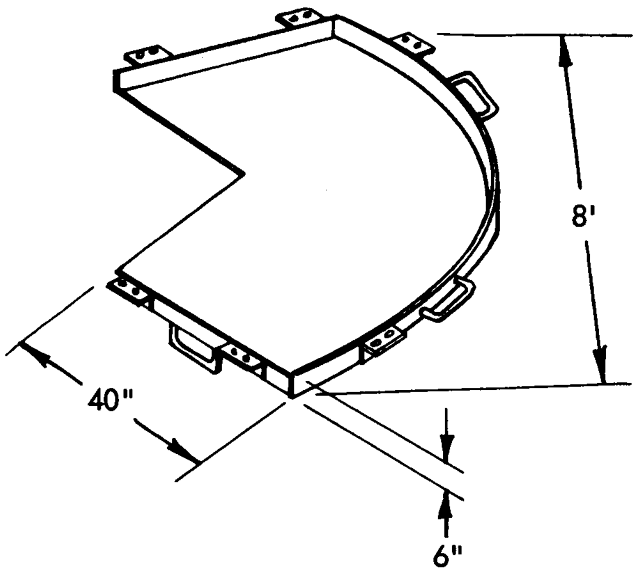


Figure 17-21. Solar Array Protective Cover, MOSE 122

Functional Requirements. The solar array covers will protect the solar arrays during handling, storage, shipment, and spacecraft integration.

Design Requirements. The protective covers must mount to the solar array structure without interference with the solar array-spacecraft interface, the solar cells, or associated MOSE. They must be rigid enough to support the load of the arrays during installation of the arrays with the solar array mounting fixture (MOSE 119). The covers must be transparent to allow for inspection and functional tests. Air vents must be provided to avoid overheating.

Description. The solar array protective covers are molded transparent plastic covers of ribbed construction. The attachment points (to the solar array structure) are reinforced with metal as required. Hand grips are provided to facilitate personnel use.

Test Requirements. The covers will be fit checked and load tested before use.

Interface Definition. The protective covers have mechanical interfaces with the solar array structure and the solar array mounting fixture (MOSE 119).

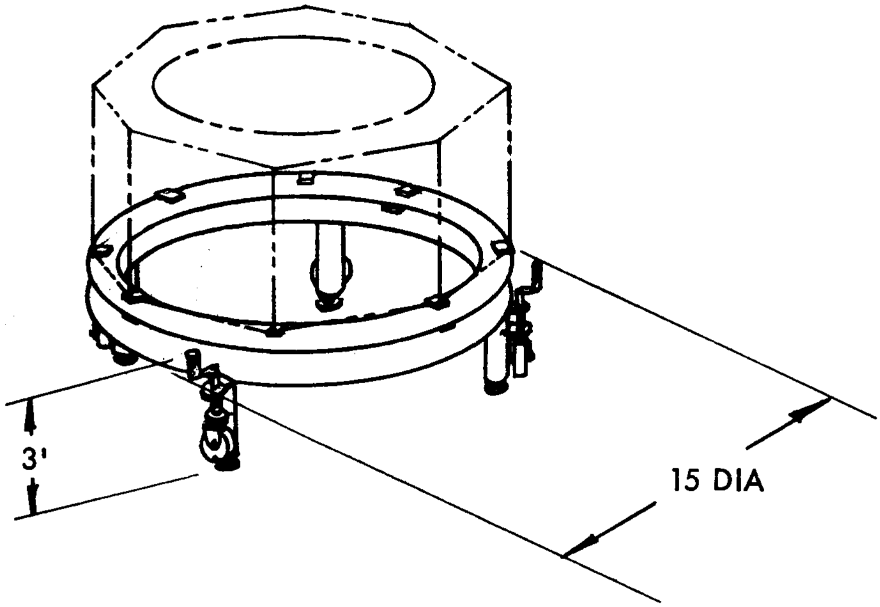


Figure 17-22. Equipment Module Mobile Stand, MOSE 128

Functional Requirements. The spacecraft equipment module must be supported and protected, within a specific facility during and between various assembly and test operations.

Design Requirements. The spacecraft equipment module mobile stand provides support and transportability for the equipment module during assembly and test operations. The stand provides physical protection around the periphery of the structure to prevent inadvertent contact with objects. It is capable of supporting a load of 3000 pounds.

Description. The mobile stand consists of a horizontal ring with four equally spaced brackets about the inner periphery. The ring is supported by four columns. The column heights can be adjusted by jack pads to provide true positioning of the spacecraft. Affixed to each column is a retractable, shock absorbing caster assembly for mobility. A removable tow bar is provided to facilitate moving the stand with equipment module to various work stations.

Test Requirements. The mobile stand will be functionally tested to demonstrate required performance and proof-load tested to demonstrate design and fabrication adequacy.

Interface Definitions. The mobile stand interfaces with the handling ring set (MOSE 1). It also interfaces with the electric prime mover (MOSE 52).

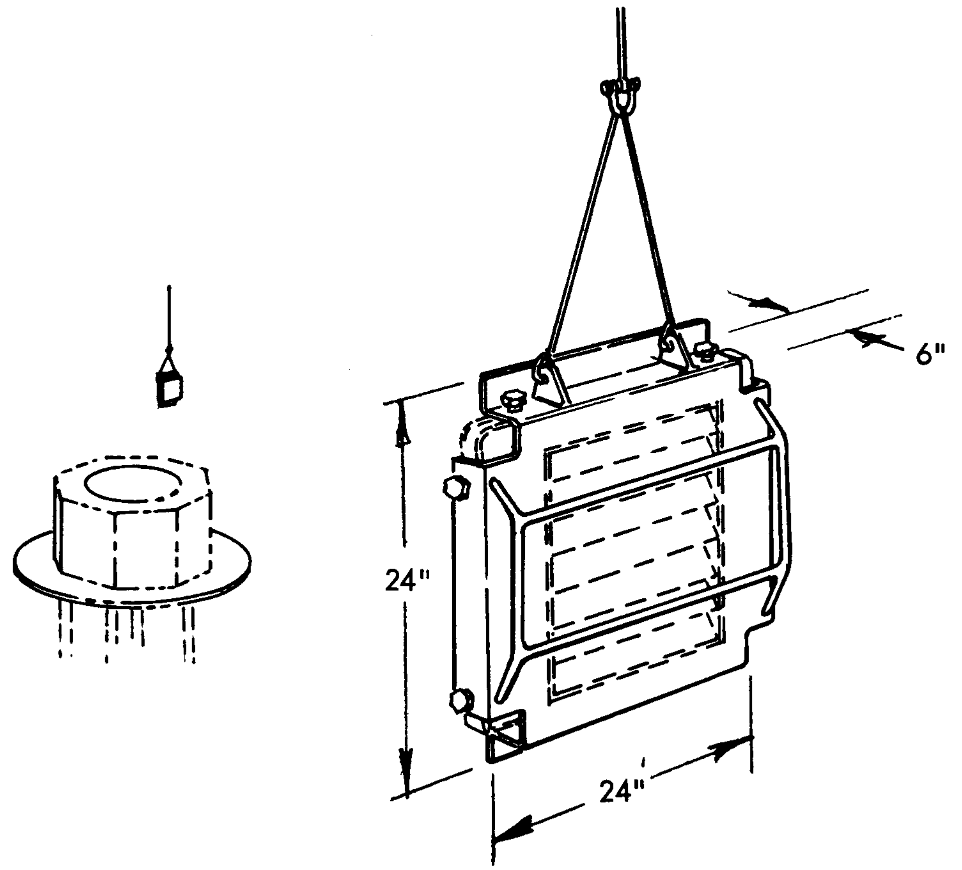


Figure 17-23. Louver Installation Device, MOSE 131

Functional Requirements. The louver installation device must provide physical support to the louver structure during testing and assembly on the spacecraft.

Design Requirements. The louver installation device must attach to the louver assembly without interference with the louver-spacecraft interface and without distorting the louver assembly structure. The louver installation fixture must provide an attachment point for a hydra-set from which the louver assembly can be suspended in its installed position while it is being attached to the spacecraft.

Description. The louver installation fixture consists of a rigid, light-weight, nonmagnetic structure with quick release attachments to the louver structure. A hoist and sling attachment point as well as carrying handles is provided. This concept, based on the Mariner units, is shown.

Test Requirements. Functional testing and load testing will be performed on the fixture before use.

Interface Definition. The louver installation fixture has mechanical interfaces with the louver assemblies and hydraset (MOSE 4).

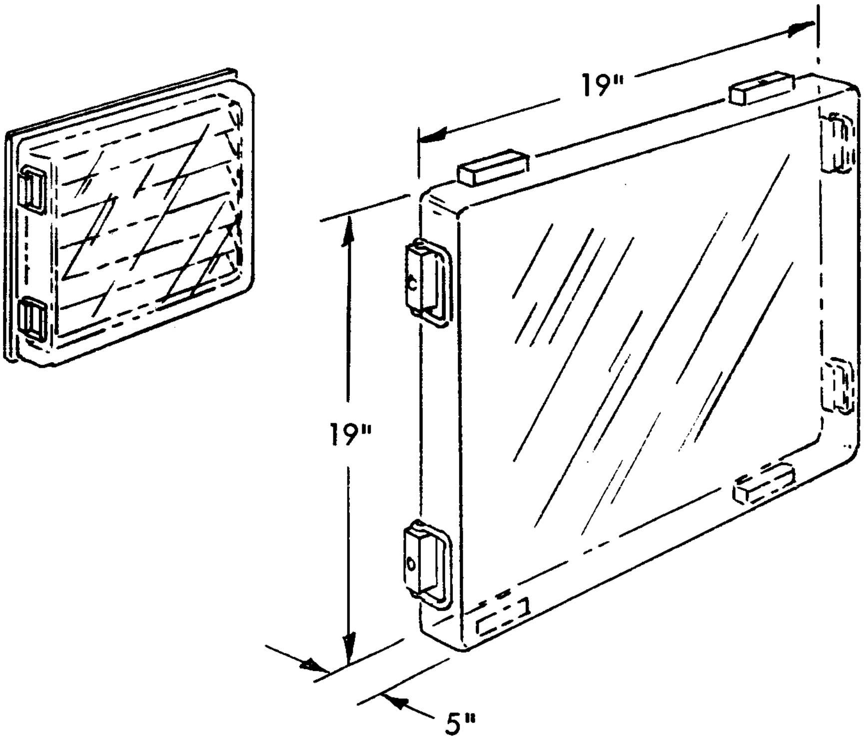


Figure 17-24. Louver Protective Cover, MOSE 132

Functional Requirement. The temperature control louver protective covers are required to protect the louvers during spacecraft assembly and test.

Design Requirement. The temperature control louver protective covers must attach, with a quick disconnect device, to the louver frame assembly without interference with the louver frame-spacecraft interface. The covers must be transparent and provide clearance for attachment of the louver installation and handling device (MOSE 131). The protective covers must not interfere with the normal operation of the louvers as they respond to temperature changes.

Description. The protective covers are 1/4-inch thick molded acrylic or other transparent plastic. Attachment points are reinforced with metal as necessary. Folding handles are provided for installation and removal of the covers.

Test Requirements. All temperature control louver protective covers will be fit checked and functionally tested.

Interface Definition. The protective covers have a mechanical interface with the louver frame assembly.

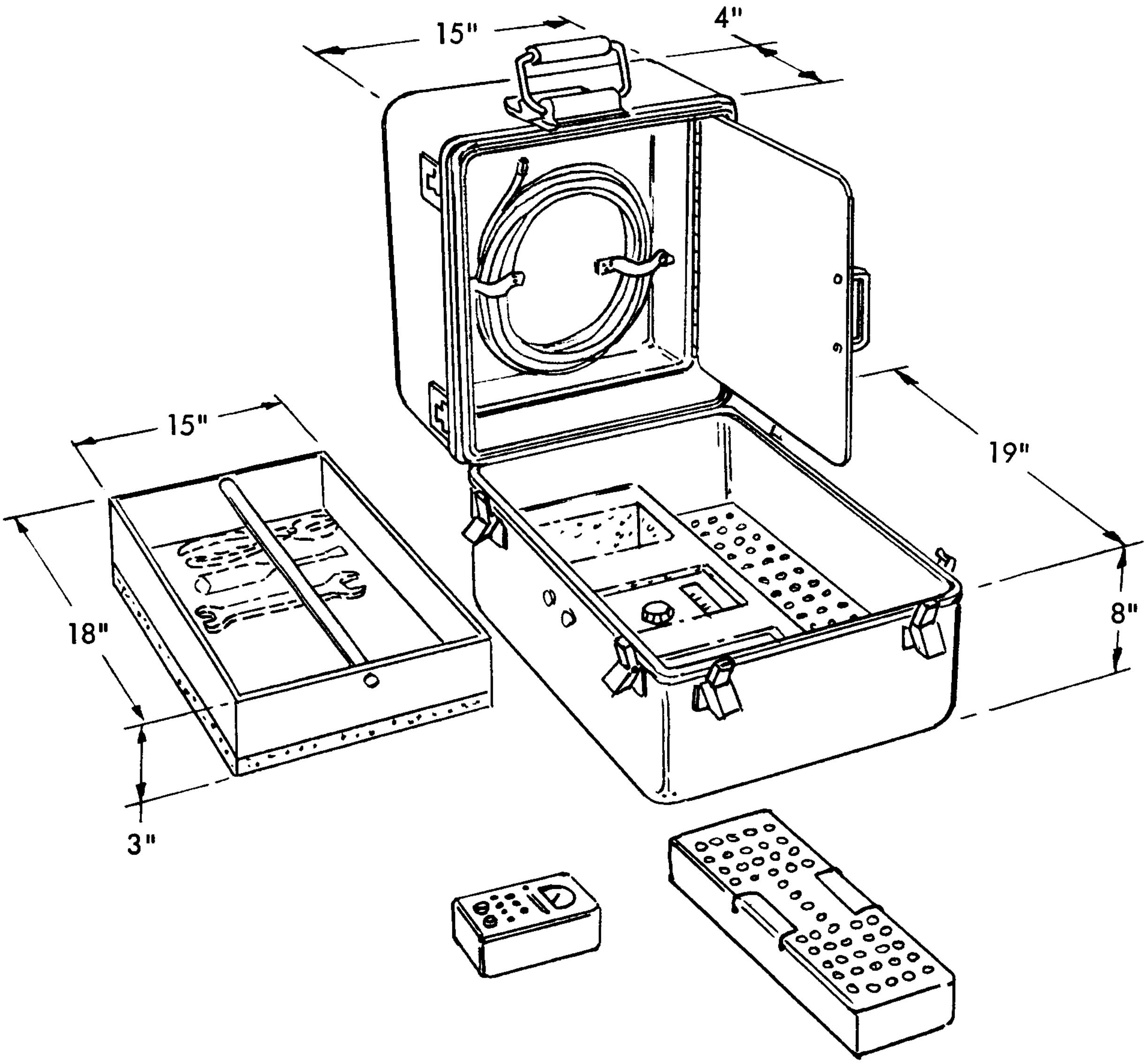


Figure 17-25. Ordnance Checkout Kit and Handling Case, MOSE 135

Functional Requirements. Positive accountability and installation and checkout capability is required for all spacecraft electro-explosives, simulators, and shorting plug devices. Means for transporting these devices and pyrotechnic circuit test instruments to and from the explosive safe area (ESA) is also required.

Design Requirements. The arming kit handling case must be portable, equipped with carrying handles, and provided with recesses for visual access to all ordnance and arming devices, test instrumentation, and tools required to test pyrotechnic circuits and to arm the spacecraft. Conductive material must be used for all components of the handling case, and the case must be provided with means for positive grounding.

Description. The handling case consists of an aluminum wiring shelf, an aluminum tooling tray, two die-cut polyurethane foam pads, two plastic trays, and a standard aluminum military transit case. The wiring shelf, of sheet aluminum, is hinged and latched to the top of the handling case. The depth of the shelf is 3 inches minimum. The tooling tray, also sheet aluminum, is provided with an aluminum handle across the entire length of the tray. A 2.4-pound density polyurethane foam pad is bonded to the bottom of the tray. A carbonized conductive polyethylene film (Velostat) is placed around each die-cut foam pad to prevent static charge buildup.

Test Requirements. The handling case will be drop-shock tested and tested to demonstrate static charge inhibition and ground continuity.

Interface Requirements. The handling case has no physical or electrical interface with other operating equipment, but provides mounting recesses for all spacecraft ordnance devices and checkout equipment.